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Technical Memorandum No. 33-52

System Considerations for the Manned Lunar-Landing Program

(NASA-CR-133660) SYSTEM CONSIDERATIONS FOR THE MANNED LUNAR LANDING PROGRAM (Jet Propulsion Lab.) 68 p

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JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

August 3, 1961

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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Technical Memorandum No. 33-52

System Considerations for the Manned Lunar-Landing Program

33

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PASADENA, CALIFORNIA

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PREFACE

In undertaking a program directed toward a manned landing on the surface of the Moon and return to the Earth, it is necessary that the National Aeronautics and Space Administration (NASA) evaluate the possible system concepts for the mission in order that the most promising system, or systems, can be selected, and that the corresponding launching-vehicle and spacecraft developments can be initiated in sufficient time. Two system concepts have received the greatest attention in the preliminary investigations conducted by NASA: (1) the use of a very large liquid-propellant multi-stage rocket *(Nova)* which carries out the entire mission in a single operation; (2) the use of somewhat smaller liquid-propellant rockets (the *Saturn*) which carry out the missions by rendezvous in a near-Earth orbit and the ensuing assembly and/or refueling operations in the free-space environment. Many alternative system concepts can be devised.

In response to an invitation from the NASA, two preliminary design studies of alternate system concepts of the manned lunar-landing and return mission have been prepared: (1) the use of solid-propellant rockets in launching-vehicle systems; (2) the use of rendezvous on the lunar surface as a method for carrying out the complete mission. (The lunar-surface rendezvous with the *Saturn C-2* launching vehicle, treated by the Lundin Committee, is summarized in "A Survey of Various Vehicle Systems for the Manned Lunar Landing Mission," NASA, June 10, 1961.) The two system concepts reported here are, of course, closely related to the two which have, up to this time, received primary emphasis. The variations in concept, however, have many inherently desirable characteristics which require evaluation prior to the time a definitive development plan is formulated for the manned lunar-landing mission. The two systems represent very different approaches to the lunar-landing mission. The lunar-rendezvous concept places the minimum requirement on the development of newer and larger launching-vehicle systems; whereas, the very large rocket places a minimum requirement on the development of an operational capability away from an Earth environment. It is concluded that the solid propellant offers a most favorable approach to the very large vehicle development in terms of development cost and time. To some extent, the system concepts are complementary in that lunar-rendezvous techniques could be used to increase the certainty of mission success with

PREFACE (Cont'd)

a vehicle system capable of carrying out the complete mission with a single launching. In addition, any concept of continuing manned-research (or other) operations on the lunar surface necessarily involves the incorporation of lunar-rendezvous techniques.

These system studies are not presented as specific proposals recommended for implementation, but rather as preliminary studies of systems having particularly favorable characteristics which, it is suggested, should be considered by the NASA in the over-all evaluation of the system, or systems, chosen for final development.

In reviewing the schedule and cost information presented in this report, it must be remembered that neither of the two studies encompasses the entire gamut of activities that must be integrated for the complete mission accomplishment. The cost estimates cover the needs of the contractors carrying out the portions of the over-all project treated in the study. The estimates do not include the general governmental administrative overhead, the Air Force range costs at Cape Canaveral, or the NASA world-wide tracking and communications costs. Both studies assume that the *Apollo* spacecraft developments are proceeding on a schedule which fits into the system being studied. The solid-propellant *Nova* study does not include cost estimates for the extensive flight-test program, using *Saturn C-1* and smaller vehicles, which is required for the spacecraft development prior to entering the *Nova* flight phase. The cost estimates for those portions of the *Apollo* spacecraft development that are included in this study are taken from the current NASA Headquarters program reviews. No attempt has been made to evaluate or to duplicate these estimates.

The two studies use different ground rules for establishing their schedules. The schedule for the lunar-surface rendezvous concept is dependent on the *Saturn C-3* development program, and a current estimate of this schedule was used as the basic timing criterion to establish the later phases of the spacecraft-system development. (See Fig. 3 of the Fleming *Ad Hoc* Task Group Summary Report, NASA, June 16, 1961.) The schedule for the solid-propellant *Nova* system was estimated as an independent system, although the possibility is noted that the spacecraft-development schedule might become the pacing item.

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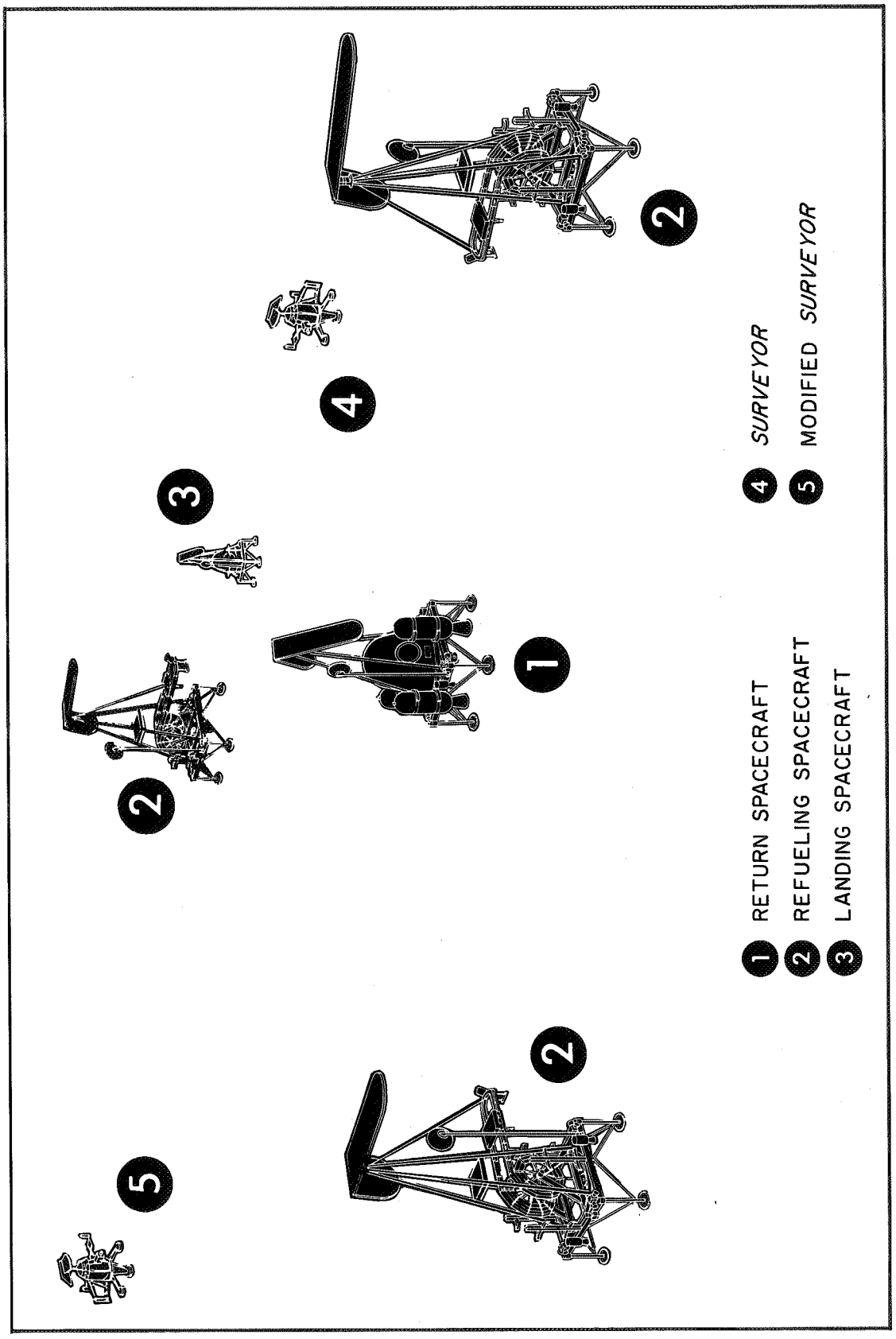

Lunar-Surface Rendezvous Concept

I. A LUNAR-SURFACE RENDEZVOUS SYSTEM

A. Introduction

The early accomplishment of the manned lunar landing and return mission has been chosen as a primary goal of the space exploration program of the National Aeronautics and Space Administration (NASA). Consequently, the several elements of the NASA have undertaken studies of a wide variety of manned lunar-landing and return mission systems and concepts. This Section presents the results of a study performed by the Jet Propulsion Laboratory (JPL) on the use of a lunar-surface rendezvous technique for mission accomplishment.

The lunar-surface rendezvous technique was chosen for detailed study because it offers several inherent advantages, both in terms of early and reliable mission accomplishment and in future growth potential. Advantages which apply to early and reliable mission accomplishment include:

1. Use of a natural body; that is, the Moon, which maintains itself, as a space station or assembly area. Weightlessness problems are avoided and known lunar conditions may be used as design criteria.
2. Mission accomplishment by means of lunar-surface rendezvous, as do other rendezvous and assembly techniques, offers flexibility in adjusting mission requirements to fluctuations in payload requirements and vehicle capability by means of adding a vehicle to the mission sequence. Thus, the critical dependence of future mission capability on decisions relating to launching vehicle design and development is reduced.
3. Ability to start parallel efforts on all system elements now, without dependence on the capability of injecting large weights into Earth-orbit or escape trajectories.
4. The automatic lunar assembly and check-out operations may be simulated on Earth.
5. Rapid reliability build-up and subsequent manrating of the system elements is accomplished by unmanned propulsion rounds which stock-pile materials on the Moon before the manned portion of the mission is carried out. The propulsion and manned payloads utilize identical cis-lunar transit, lunar landing, and lunar take-off systems.
6. The manned portion of the mission need not be committed until a checked-out return system is available on the Moon.
7. Previously landed spacecraft can provide approach guidance beacons and TV monitors for the lunar manned-landing sequence. These aids provide safety margins in the final manned-landing sequence which are comparable to those in use by Earth aircraft.
8. The propaganda value of a step-by-step lunar assembly program cannot be over-emphasized as compared to a one-shot lunar landing, even though both are accomplished at the same time.

In terms of growth potential, the lunar-surface rendezvous concept is attractive because it offers early development of techniques required by advanced missions. Extensive manned lunar exploration and the establishment of a lunar base or scientific station require the ability to perform lunar-surface operations, including the rendezvous of payloads on the Moon and the construction and maintenance of long-lifetime lunar facilities.

The study results include the following items:

1. Development of a system concept for the manned lunar-landing and return mission by means of lunar-surface rendezvous.
2. Functional descriptions and weight estimates for system elements based on mission accomplishment utilizing the C-3 launching vehicle.

3. An operational sequence for the mission.
4. System variations for a three-man mission and a minimum one-man mission.
5. Examination of the feasibility of two critical mission functions; that is, the lunar-landing accuracy and the lunar-surface operations.
6. A review of the program leading up to the manned lunar-landing and return mission.
7. Cost and schedule data for mission accomplishment.

The following ground rules and mission-design criteria have governed the study effort:

1. Mission accomplishment as soon as possible.
2. Maximum reliability for all manned systems and operations.
3. Demonstration of system performance and reliability before man is included in the system.
4. Compatibility of system elements (booster, spacecraft, ground facilities, and a complete monitoring scheme) as to schedules, problem areas, et cetera.
5. Human factors and manned support-system development and re-entry technology program to be carried out in near-Earth space before the manned lunar mission is performed.
6. Development of man's capabilities in both the space and lunar environment is a major program goal. The system must be capable of completely automatic operation and must provide over-ride capability when desirable in order that man will have maximum flexibility in adapting his tasks to the situation in which he finds himself.

B. Mission Concept

In considering the system required to perform a mission as complex as the manned lunar landing and return, it is best, where possible, to divide the system into a finite number of parts and examine the technical feasibility and state of the art for each part or module. This study approach allows the concentration of the appropriate skills on each problem and permits early identification of the critical or pacing portions of the system. An over-all mission concept, which permits mission accomplishment and avoids as many problem areas as is possible, may then be developed by considering the several system modules.

The modules of the system may be defined either by function or by any other appropriate criteria as long as the relationship, or interface, between modules is kept as simple as possible. If the various interfaces between modules within the system become complex and difficult to define, the advantages of the modular approach are lost and system integration becomes the pacing item of the project.

An appropriate set of modules for the manned lunar-landing and return system may be chosen by identifying the functions required by the mission. The broad functional requirements for successful mission accomplishment are as follows:

1. Escape from Earth
2. Transit to the Moon
3. Lunar landing
4. Lunar-surface operations
5. Lunar launch and injection
6. Return transit to Earth
7. Re-entry into the Earth's atmosphere
8. Recovery of the re-entry body
9. Support and protection of human crew members throughout all phases of the mission
10. Abort capability for the manned portion of the system.

Grouping of related functions results in the following set of gross system elements:

1. Command module (re-entry body)
2. Mission module
3. Transit, landing, and return module
4. Earth injection system (launching vehicles and rendezvous and assembly techniques)

The characteristics of each module are discussed in the following paragraphs.

1. Module Characteristics

Command Module. This module includes human crew and support and protection for the crew. Because of the abort requirement, the unit must also be the re-entry body. This module must contain life support for a time period sufficient to cover both normal and abort modes. Food and other supplies for extended lunar operations may be stored in other parts of the system. The primary items affecting the design of this module are human factors and re-entry conditions.

Mission Module. This package provides additional human supplies for extended mission lifetime (food, etc.)

and the instruments and equipment required by the particular mission. The unit may be environmentally controlled in order to provide additional operating area for the crew.

Transit, Landing, and Return Module (the Bus). This unit is a transportation spacecraft capable of performing the following functions:

a. Transportation of a payload from near-Earth injection to the vicinity of the Moon (including midcourse corrections).

b. Precise soft landing of the payload on the lunar surface (including hovering and translation capabilities).

c. Launch of the Earth-return system from the lunar surface.

d. Transportation of a payload from the vicinity of the Moon to the selected Earth re-entry corridor (including midcourse corrections and terminal maneuver).

All of the spacecraft subsystems (such as attitude control, power supply, propulsion, landing structure, etc.) required to achieve these functions are included in this bus module and are distinct from the payload; that is, the bus can perform its mission, regardless of the payload, for any payload meeting the gross restriction of weight, volume, shape, center of gravity, location, et cetera. The guidance system employed by the bus would include an override mode in order that direction of the final landing sequence could be accomplished either by an Earth-command link or by the crew. The bus also provides certain support functions to the payload in the form of power supply, communications capability, and available fields of view.

Earth Injection System. The launching vehicle required to perform the manned lunar-landing and return mission is dependent on two factors: (1) the weight of the several mission elements; (2) the energy management scheme employed. The several alternate approaches for the mission of interest include:

a. Direct launch of the complete mission system by a *Nova*- class booster.

b. Launch of system elements by a *C*-3- class booster, with system assembly being performed

(1) In Earth orbit

(2) In lunar orbit

(3) On the lunar surface, or

(4) Various combinations of (1), (2), and (3).

2. Selected Mission Concept

The particular mission concept discussed in detail in the following pages would require the accomplishment of the initial manned lunar mission with *Saturn C*-3 launched spacecraft assembled on the lunar surface. A *C*-3 launched spacecraft can soft-land approximately 10,000 lb on the lunar surface. Previous *Surveyor* or *Ranger* spacecraft would establish the landing site and provide homing beacons, TV monitoring equipment, et cetera. A number of methods for refueling on the lunar surface may be envisioned.

One possible concept follows: Three 10,000 lb refueling spacecraft would be landed at points approximately equally spaced around, and within 45 feet of, a spacecraft comprised of a bus and unmanned command module suitable for returning three men to Earth. Solid-propulsion units would be transferred from the refueling spacecraft to the centrally located return spacecraft by means of specially designed transfer tracks. The assembly operation would be monitored by TV, and the assembled vehicle would be checked-out before sending a man from Earth to the area via a second spacecraft comprised of a bus and command module.

The lunar complex so established would be capable of maintaining itself in the lunar environment. The crew would perform any required lunar mission functions upon landing and would then transfer to the waiting return vehicle and return to Earth. If, for any reason, the crew should determine that the previously landed and assembled spacecraft should not be used for the return flight, the automatic refueling equipment could be used to retransfer the propulsion units to the spacecraft in which the crew arrived; this spacecraft would then be used for the return to Earth. The three solid-propellant rockets used for launch from the lunar surface are identical with the retro rockets used for each vehicle in landing on the Moon. These retro rockets are jettisoned at some distance from the assembly site during the vernier descent and touchdown maneuver. Thrust-vector control and propulsion for the vernier descent and touchdown is supplied by throttlable liquid vernier motors. A large amount of the technology developed for *Surveyor* would be utilized in this concept for the manned lunar landing. The return-vehicle weight would be approximately 29,800 lb at lunar takeoff, 10,700 lb at lunar injection, 10,000 lb as the Earth is approached, and 5300 lb at Earth entry. These weight performances are based on scaled-*Surveyor* and *Apollo* technologies as indicated by NASA and contractor studies.

The *Saturn* C-3 would be used throughout for the Earth-based launching vehicle. A minimum of 5 successful launches would be required for the basic mission. The actual number required to accomplish the mission would be a direct function of the success rate of the firings and assembly operations on the lunar surface; however, any failure before the manned landing would not affect the success of the manned lunar-landing sequence. Identical transit, landing, and return modules would be used in all launches and landings; only the payloads would differ; that is, command modules or return propulsion.

The abort capabilities provided by this mission concept are for the return of the manned command module to Earth for as large a portion of the mission as is possible. A separable abort rocket system would perform the abort function during the atmospheric portion of the flight and functions in the event of a catastrophic failure of the booster vehicle. A penalty of 250 lb to the injected payload would occur in maintaining this capability up to jettisoning of the unit (prior to injection). In the event that the mission were to be aborted after the jettisoning and prior to the lunar terminal phase, the lunar retropropulsion and vernier-propulsion units would be utilized to abort.

Detailed descriptions of the system elements and the mission steps are given in subsequent paragraphs.

C. *Program Aspects*

A brief examination of the lunar exploration program leading up to the manned lunar-landing and return mission, and eventually to the establishment of a lunar base or scientific observatory, has been included in the present study. The interrelationship of the various developments and events leading up to the accomplishment of the manned lunar-landing and return mission may be demonstrated by examining the mission requirements. Gross requirements for the capability to perform the manned lunar-landing and return mission are:

1. Determination of the data required for system design. The primary area of interest is the definition of the several environments – cis-lunar, lunar, and re-entry – in which the system must operate and provide protection and support for the human crew. Several experimental programs are required to obtain such data.
2. Determination and development of the techniques required for mission accomplishment. The primary areas of technology required by the manned lunar-landing and return mission are:
   a. Launching vehicle and energy-management technology (including rendezvous techniques)
   b. Space-transit technology
   c. Vacuum-landing technology
   d. Human factors and manned support-system technology
   e. Re-entry and recovery technology.
3. Development of the actual system hardware, both flight hardware and ground-support hardware and facilities.

In order to fulfill these requirements, the elements of the several NASA space exploration programs will be utilized. Figures 1 through 6 show the growth in environment definition and development of technology. The hardware development schedule for the actual mission is shown in Part I of this Section.

D. *System Elements; Three-Man Lunar-Landing Mission*

The three-man lunar-landing mission contains the system elements listed below, in addition to the *C-3* launching vehicles. The spacecraft must contain the spacecraft bus module and may contain various other modules.

1. Command module
2. Spacecraft bus module
3. Environmental control support capsule
4. Mission module
5. Propulsion module

Figure 7 is a diagram showing the relationship of the various system elements.

1. Command Module (Manned Capsule)

This module has the following functions: life support; mission parameter displays for crew; provides for manned participation in the flight and evaluation of man's capability in space; couples man with the communication system; provides the Earth entry, landing, and recovery capability; and, to a major extent, participates in the abort modes of operation.

The module presented here is derived from the *Apollo* Industry Study Capsules, considering the following modifications to meet the lunar rendezvous mission concept:

The basic space-abort and lunar-return unit is composed of the command module and the bus. Therefore, all life-support systems must be on these modules with the major portion on the command module.

PROJECT	DATA	CALENDAR YEAR 1961	1962	1963	1964	1965	1966	1967
SOLAR OBSERVATORY SATELLITES	SOLAR-ACTIVITY MEASUREMENTS							
SOUNDING ROCKETS *NIKE-CAJUN* *JOURNEYMAN*	RADIATION DATA DURING SOLAR FLARES	AVAILABLE ON STANDBY BASIS						
RANGER 1, 2, 10, 11	SPACE ENVIRONMENT							
MARINER A, B	SPACE ENVIRONMENT							
VOYAGER	SPACE ENVIRONMENT							
RANGER 3-9	LUNAR ENVIRONMENT							
SURVEYOR (LANDER AND ORBITER)	LUNAR-ENVIRONMENT AND LANDING-SITE SURVEYS							
PROSPECTOR	LUNAR-ENVIRONMENT AND LANDING-SITE SURVEYS							

Figure 1. *Cis-lunar and lunar environmental data*

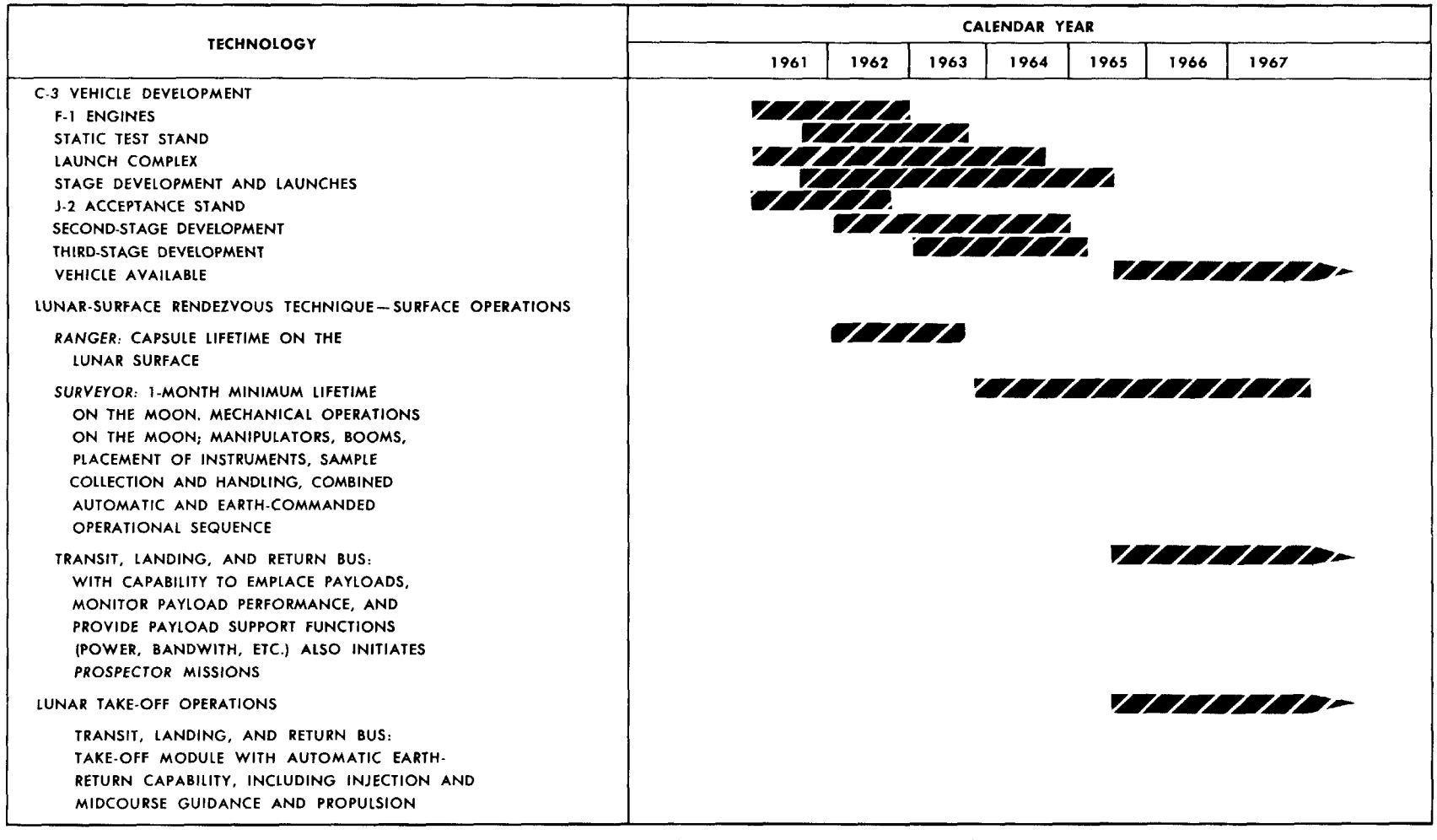
TECHNOLOGY
CALENDAR YEAR
1961
1962
1963
1964
1965
1966
1967
C-3 VEHICLE DEVELOPMENT
F-1 ENGINES
STATIC TEST STAND
LAUNCH COMPLEX
STAGE DEVELOPMENT AND LAUNCHES
J-2 ACCEPTANCE STAND
SECOND-STAGE DEVELOPMENT
THIRD-STAGE DEVELOPMENT
VEHICLE AVAILABLE
LUNAR-SURFACE RENDEZVOUS TECHNIQUE—SURFACE OPERATIONS
RANGER: CAPSULE LIFETIME ON THE LUNAR SURFACE
SURVEYOR: 1-MONTH MINIMUM LIFETIME ON THE MOON. MECHANICAL OPERATIONS ON THE MOON; MANIPULATORS, BOOMS, PLACEMENT OF INSTRUMENTS, SAMPLE COLLECTION AND HANDLING, COMBINED AUTOMATIC AND EARTH-COMMANDED OPERATIONAL SEQUENCE
TRANSIT, LANDING, AND RETURN BUS: WITH CAPABILITY TO EMPLACE PAYLOADS, MONITOR PAYLOAD PERFORMANCE, AND PROVIDE PAYLOAD SUPPORT FUNCTIONS (POWER, BANDWITH, ETC.) ALSO INITIATES PROSPECTOR MISSIONS
LUNAR TAKE-OFF OPERATIONS
TRANSIT, LANDING, AND RETURN BUS: TAKE-OFF MODULE WITH AUTOMATIC EARTH-RETURN CAPABILITY, INCLUDING INJECTION AND MIDCOURSE GUIDANCE AND PROPULSION

Figure 2. Launching-vehicle and lunar-operations technology

PROJECT	CALENDAR YEAR 1961 1962 1963 1964 1965 1966 1967
RANGER	
MARINER A, B	
SURVEYOR	
PROSPECTOR	
VOYAGER	

Figure 3. Space transit technology

PROJECT	TERMINAL PHASE TECHNIQUE	CALENDAR YEAR 1961 1962 1963 1964 1965 1966 1967
RANGER	ROUGH LANDING, OMNIDIRECTIONAL, 3000 EARTH G LANDING SHOCK CPE = 100 KM (3σ)	
SURVEYOR	ATTITUDE-CONTROLLED LANDING, RADAR-DOPPLER GUIDANCE AND CONTROL LOOP WITH MAIN AND VERNIER RETRO SYSTEMS. 100 EARTH G LANDING SHOCK. EARLY *SURVEYORS*: CPE = 75 KM (3σ) LATER *SURVEYORS*: CPE = 10 KM (3σ) (ADDITION OF RANGING EQUIPMENT)	
TRANSIT, LANDING, AND RETURN BUS	*SURVEYOR* TECHNOLOGY WITH ADDITION OF A "HOMING" SYSTEM (USES LUNAR BEACON) COMMAND OVER-RIDE SYSTEM FOR EITHER EARTH OR S/C COMMAND SYSTEM. SITE SELECTION OF ± 5 METERS WITH RESPECT TO PREVIOUSLY LANDED S/C. MAXIMUM LANDING SHOCK OF 50 EARTH G'S. MAXIMUM SUSTAINED ACCELERATION LIMITED TO 10 EARTH G'S OR LESS.	

Figure 4. Vacuum-landing technology

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PROJECT	DATA
BALLOON FLIGHTS	ANIMAL, RADIATION, COSMIC PRIMARIES
NERV PROGRAM	ANIMAL, RADIATION, INNER-BELT PROTONS
MERCURY	
5-MIN FLIGHT	ANIMAL AND MAN; WEIGHTLESSNESS, HUMAN-SUPPORT SYSTEM
ORBIT: 4.5 HR	ANIMAL AND MAN; WEIGHTLESSNESS, HUMAN-SUPPORT SYSTEM
ORBIT: 27 HR	ANIMAL AND MAN; WEIGHTLESSNESS, HUMAN-SUPPORT SYSTEM
ORBIT: 2 TO 14 DAYS	ANIMAL ONLY, EXTENDED LIFETIME
APOLLO	
ORBIT: 2 TO 14 DAYS	MAN, EXTENDED LIFETIME

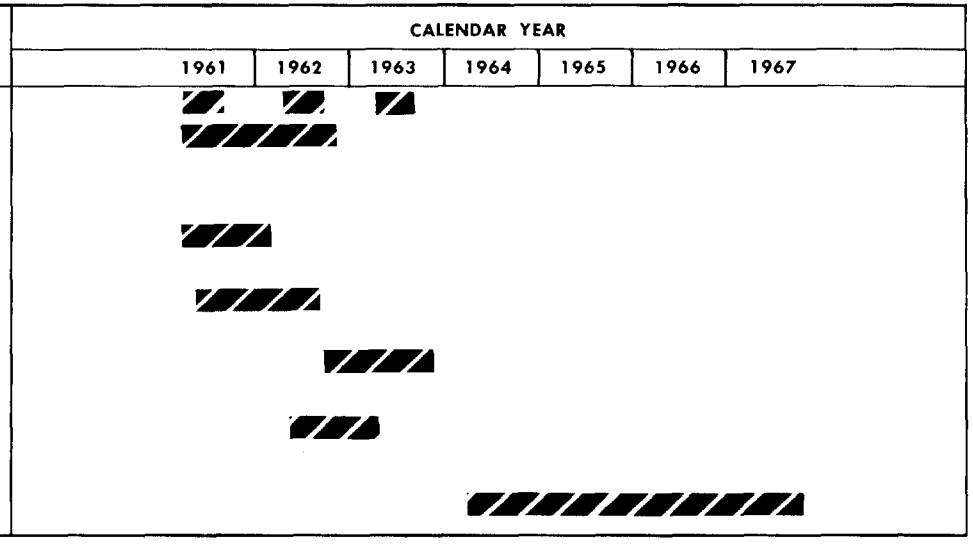

Figure 5. Human factors and manned support-system technology

PROJECT	DATA
MERCURY	ELLIPTICAL RE-ENTRY
AGENA RE-ENTRY TEST	PARABOLIC RE-ENTRY OF 1/3-TO 1/2-SCALE *APOLLO* COMMAND MODULE. AEROTHERMODYNAMIC AND MATERIALS EVALUATION
APOLLO	ELLIPTICAL AND PARABOLIC RE-ENTRY OF MANNED COMMAND MODULE

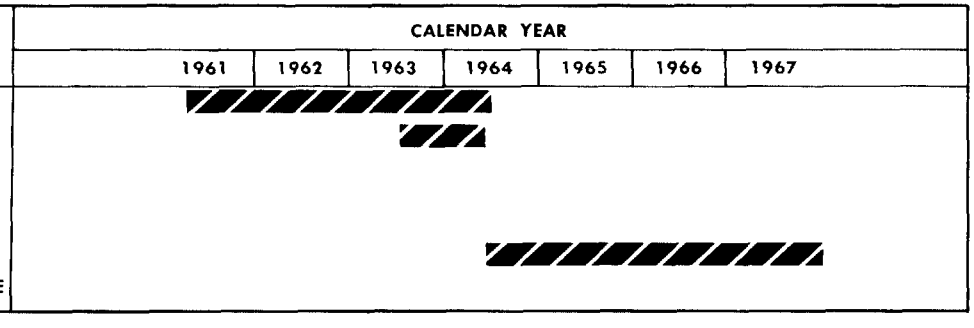

Figure 6. Re-entry and recovery technology

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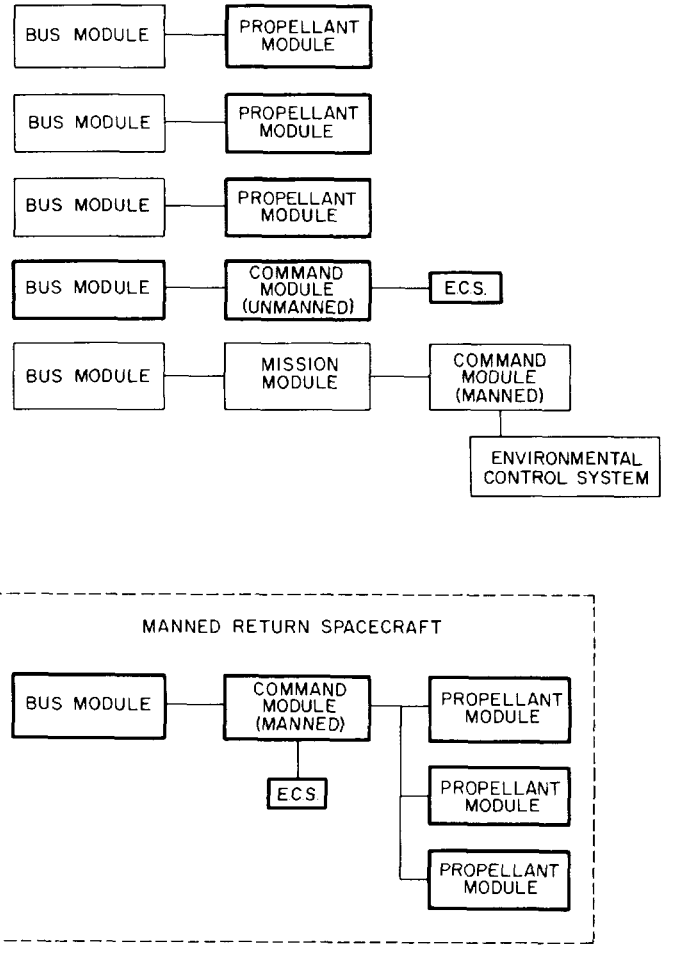

Figure 7. Manned lunar-landing mission concept

No radiation protection other than that afforded by the structure and the equipment has been specifically included in the command module. However, the mission weight summaries (Tables 1 through 8) show a weight contingency allowance of approximately 750 lb. This weight might be used to provide radiation shielding in areas where the command module structure is insufficient for human protection. A decision in this area depends on further definition of the radiation shielding requirements and on the actual design of the command module. In the event that radiation data obtained in the future indicate a heavy shielding requirement, the mission may be modified to include a heavier command module and additional refueling payloads.

Primary items affecting command-module design are:

a. Entry and recovery mission.

b. Space environment, including radiation and meteorite protection, and weightlessness.

c. Required lifetime, including normal operations and abort modes.

d. Methods of manned participation in the mission and study of man's capability.

e. Number in crew.

f. Provision for use in other missions; training, Earth-orbital flights, direct use in manned Earth rendezvous, et cetera.

2. Spacecraft Bus Module

The spacecraft bus performs the cis-lunar transit, lunar landing, lunar launch, and return portions of the mission. The techniques performed by the bus system described in this study are closely related to those of the *Ranger* and *Surveyor* spacecraft systems. Certain additional functions have been included which are directly related to the mission of interest. In particular, the capability for extended hover and translation modes and the ability to land at a specific point on the lunar surface are included in the proposed bus system. Weight allowances include provisions for redundancy in appropriate portions of the system. The bus system recommended here is essentially that required for the transportation and landing of the *Prospector* scientific packages.

The bus module described in the weight summary for the three-man lunar-landing mission (Table 4) is based on nominal *C-3* injection capability of 30,000 lb. The lunar-landing and takeoff propulsion weights are those for storable solid-rocket systems ($I_{sp} \sim 300$ sec); this results in the capability of soft-landing approximately one-third of the weight injected to the Moon and of launching toward the Earth approximately one-third of the gross lunar takeoff weight. Identical propulsion units are used for both lunar landing and takeoff sequences. It is believed that this approach to the lunar propulsion units (i.e., moderate performance requirements and maximum reliability buildup by heavy use-rate of the unit) provides the best solution to the problem of the lack of abort capability during the lunar-launching phase of the manned mission.

As the performance of the *C-3* launching vehicle is not firm at this time, this study includes an examination of weight variations in the spacecraft bus and final landed lunar payload as a function of injection weight. The expected bus and lunar payload weights for a range of injection weights covering both the *C-2* and *C-3* vehicle capabilities (10,000 to 45,000 lb) are shown in Table 8 of the weight summary Tables.

Table 1. Three-man mission concept

Step	Spacecraft	Booster	Mission
1	*Ranger* and/or *Surveyor*	A-A	Select landing site and gather knowledge on the site.
2	*Surveyor*	A-C	Establish radio "homing" beacon on-site. Establish visual monitoring capability on-site.
3	Bus + propulsion payload	C-3	Cache return propulsion on lunar surface (minimum of 3 firings).
4	Bus + command module (unmanned) +E.S.C. package + vernier propellant unit	C-3	Cache return S/C for crew on lunar surface. Cache vernier propellant for return transit maneuver into entry corridor.
5	– – – – – – –	– – –	Automatic fueling of return S/C (Step 4) with propulsion modules landed (Step 3). Complete system checkout.
6	Bus + command module (manned) + E.C.S. package + mission module	C-3	Carry three-man crew to lunar cache.
7	– – – – – – –	– – –	Checkout of return S/C established in Step 5 in preparation for return to Earth. (Note: In case the system established in Step 5 is not O.K., the crew may replace selected subsystem or have return propulsion transferred to their own landed S/C for return.)
8	Bus + command module (manned) + E.C.S. package + extra vernier propellant unit	Lunar-assembled return system	Lunar launch, return transit and re-entry, and recovery of crew.

Note: Figure 7 is a diagram showing the relationship of the various system elements.

3. Environmental Control Support Capsule (E.C.S. Package)

This module is called out separately in order to emphasize the mission concept, and the unit may or may not be a physically separable item from the command module.

It is evident from the *Apollo* studies and from the lunar assembly mission concept that some units of the required life support are essential for the space mode of operation but not essential to the entry function. In the *Apollo* studies, these items are, in general, integrated into the mission modules, but, for the lunar assembly concept, they must attach to the bus or command module. Functions covered are:

a. Parts of the life-support systems external to the life-support pressure vessel (i.e., radiator for thermal-control system).

b. Redundancy required for space operations but not for entry and recovery (i.e., spare atmosphere-control system as required).

c. Provision of added life-support capacity for mission module (i.e., additional gas-charge storage).

d. Augmentation of basic bus functions (i.e., power for life-support operations).

The interfaces for this unit will be highly dependent on the equipment included in it, but a basic guideline is that it be jettisoned with the bus before entry.

4. Mission Module

The concept proposed here presents the mission module as an easily interchangeable unit with a minimum interface with the rest of the system. The major point of interest is that it can be attached as a habitable work area, but it does not contain elements of the life-support system; rather, the outputs of the life-support system are supplied to it through the interface.

The specific module proposed here is carried on the manned Earth–Moon transit. It provides three major functions:

a. A habitable area of volume equal to that of the command module during the Earth–lunar transit and the landed operations.

b. A base for Earth–lunar transit and landed manned science operations.

c. A base for the manned parts of the lunar operations and preparation for the return transit.

Table 2. Over-all weight summary for three-man mission

Spacecraft			Weight, lb									
Description	Weight, lb		At Liftoff	At Injection	Prior to Main Retro	Completion of Main Retro	Touch-down	Lunar Payload	Prior to Lunar Takeoff	At Injection from Moon	In Near-Earth Vicinity	At Re-entry
	Item	Total										
1. Propulsion S/C			30,000	30,000	29,774	10,966	10,364	7,884	....	....	....	....
Bus	2,476											
Propulsion payload[a]	7,625											
Weight contingency	263	10,364										
(3 S/C required, no abort propulsion required)												
2. Command module, S/C, unmanned			30,000	30,000	29,774	10,966	10,364	7,884	....	....	....	....
Bus	2,476											
Command module[b]	4,728											
E.S.C. package	330											
Vernier propulsion unit[c]	2,598											
Weight contingency	232	10,364										
(1 required, no abort propulsion required)												
3. Command module, S/C manned[d]			30,000	29,750	29,524	10,716	10,114	7,638	....	....	....	....
Bus	2,476											
Command module	5,268											
E.S.C. package	330											
Mission module	1,290											
Weight contingency	750	10,114										
1 required, abort propulsion required)												
4. Return S/C, manned			....	....	....	....	....	....	29,750	10,716	10,114	5,268
Bus	2,476											
Command module	5,268											
E.S.C. package	330											
Additional vernier propulsion[e] (for maneuvering to re-entry corridor)	1,290											
Weight contingency	750	10,114										
(1 required)												

[a]Replacement propulsion weight is 6130 lb.

[b]No crew results in 540-lb weight reduction.

[c]Excess vernier propulsion to be used for return maneuver to desired re-entry corridor.

[d]Abort propulsion-weight penalty at injection = 250 lb.

[e]A preliminary estimate which will depend on selected return trajectory and lunar launching site.

Table 3. Weight summary for three-man command module (capsule)

Item	Weight, lb		Remarks
	Item	Total	
1. Structure			All items are based on a review of the final *Apollo* study reports.
Heat shield	1500		
Airframe	1000		
Aerodynamic surfaces (flaps)	350	2850	
2. Communications			Communications associated with manned system and those telecommunications for near-Earth usage (boost, re-entry).
VHF system (2 complete systems; transmitters, receiver, antennas, etc.)	20		
Data recorder	25		
PCM multiplexer	38	83	
3. Guidance and control			Only those guidance and control elements associated with re-entry are carried in the command module.
Aerodynamic flight control	100		
Reaction controls	40		
Radar altimeter	50		
Miscellaneous	25	215	
4. Power system			Power supply for boost and re-entry phases only.
Batteries and power regulation	75	75	
5. Cabling and connectors	35	35	
6. Life and life-support			
Crew	540		
Environmental control system	340		
Furnishings and equipment	530		
Instruments and displays	150	1560	
7. Search and recovery			
Landing equipment (parachutes, shock absorber, flotation, etc.)	375		
Search and recovery aids (beacon, Sofar bomb, radar chaff)	75	450	
Three-man command module		5268	

The module is not returned under the present scheme but is left intact on the lunar surface. It can readily be adapted as an element of a permanent lunar base, if such is planned for the future. It seems evident that this disposition of the module contributes more to the over-all aims of the program than the alternative of including it in the return package.

The weight allocation is adequate for future alternate modules. Examples are:

a. All or part of a manned roving vehicle;

b. An equipment package for landed science operations or lunar base construction.

5. Propulsion Module

The propulsion module is one of the two payloads carried to the lunar surface by the bus. The module consists of a portion of the return propulsion, the appropriate automatic refueling equipment and refueling monitoring equipment, the environmental control equipment required to store the propulsion properly, and appropriate instrumentation to monitor the condition of the return propulsion.

E. System Elements; Minimum One-Man Lunar-Landing Mission

For comparison with the three-man mission, a one-man mission has been studied under the same general ground

Table 4. Weight summary for transit, landing, and return bus[a]

Item	Weight, lb			Remarks
	Item	Total Subsystem	Total	
Weight of S/C on pad			30,000	
Abort propulsion (effective injection weight)			250	Based on final *Apollo* study reports.
Bus				
1. Structure				Based on HAC *Surveyor* studies and scaled for very soft landing.
Spaceframe assembly	518			
Landing-gear assembly	435			
Environmental control	140	1,093		
2. Communications				Provides all tracking and command as required for transit, landing and return transit. Based on HAC *Surveyor* technology, compatible with *Apollo* studies.
S-Band system (2 complete systems; transmitetr, receiver, monopole and parabolic antennas, etc.)	27			
Command decoders	5			
Signal processors	8	40		
3. Guidance and control				Based on HAC *Surveyor* technology.
Sensors, inertial references, computer electronics, etc.	150			
Attitude-control system (valves, actuators, fittings, etc.)	275	425		
4. Power system				Provides power for *all phases of mission except capsule* requirements in boost and re-entry phases. Based on HAC *Surveyor* technology.
Solar array and mechanism	75			
Batteries and power regulator	300	375		
5. Cabling and connections	75	75		
6. Telemetry and instrumentation				Provides T/M for monitor of performance of bus, payload and external environment.
Instrumentation	40			
T/M	20	60		
7. Propulsion (non-jettisoned)				Based on HAC *Surveyor* technology. Allows for added mission hover and translation requirements.
Vernier and midcourse inert weight	408	408		
8. "Dry" bus at touchdown (Σ of items 1 through 7)			2,476	
Propulsion				
1. Vernier propellants				Based on HAC *Surveyor* technology.
Midcourse	226			
Attitude-control during main retro	418			
Hover and translation	602	1,246		
2. Main retro rocket				Based on HAC *Surveyor* technology.
Inert weight	736			
Propellants	17,654	18,390		
3. *Total expended propulsion*			19,636	
Net Lunar Payload (Injection — bus — propellant)			7,638	

[a]*Applies to both three-man and minimum one-man missions.*

Table 5. Weight summary for three-man mission environmental control support package

Item	Weight, lb		Remarks
	Item	Total	
1. Auxiliary environmental control equipment			Based on final *Apollo* study reports.
Heat control	50		
Air circulation	30		
Humidity control	5		
O_2 and N_2 control	60		
CO_2 control	30		
Gas charge	100	275	
2. Structure and packaging	55	55	
Package		330	

Table 6. Weight summary for three-man mission module

Item	Weight, lb		Remarks
	Item	Total	
1. Structure (module structure and hatches, etc.)	1000	1000	Based on final *Apollo* study reports.
2. Scientific equipment and supporting structure, power, etc.	200	200	
3. Environmental control	50	50	
4. Furnishings and equipment	40	40	
Mission module		1290	

Table 7. Weight summary for three-man mission propulsion payload (3 required)

Item	Weight, lb		Remarks
	Item	Total	
1. Propulsion module for refueling			Three modules required to refuel a landed bus/capsule S/C
Main booster			
Inert weight	245		
Propellants	5885	6130	
2. Refueling equipment			
Propulsion-transfer equipment (hoist, track, actuators, etc.)	1125		
Monitoring equipment (TV monitor)	30	1155	Utilizes bus power and rf link
3. Environmental control			
Container, etc.	300		
Monitoring T/M	40	340	Utilizes bus power and rf link
Propulsion payload		7625	

Table 8. Bus and lunar payload weights for various injection weights

Item	Weight, lb							
Weight injected toward the Moon	10,000	15,000	20,000	25,000	30,000	35,000	40,000	45,000
Bus								
Communications	40	40	40	40	40	40	40	40
Power system	375	375	375	375	375	375	375	375
Telemetry and instrumentation	60	60	60	60	60	60	60	60
Guidance and control								
Sensors, inertial reference, computer, electronics, etc.	150	150	150	150	150	150	150	150
Attitude-control system	108	150	192	234	275	317	360	402
Structure; spaceframe, landing gear, environmental control	577	705	833	964	1,093	1,221	1,351	1,480
Cabling and connections	42	50	58	67	75	83	92	100
Propulsion (non-jettisoned); vernier and midcourse inert weight	137	206	275	343	412	480	549	618
Dry bus at touchdown	1,489	1,741	1,983	2,233	2,480	2,724	2,977	3,225
Vernier propellant for hovering and translation	202	304	405	506	607	708	809	911
Main retro propulsion								
Vernier propellants	141	211	281	351	422	492	562	632
Main retro-rocket; inert weight	247	371	495	618	742	865	989	1,113
Propellants	5,934	8,901	11,868	14,835	17,802	20,769	23,736	26,703
Midcourse propellant	76	114	152	190	228	266	305	343
Total at touchdown	3,400	5,099	6,799	8,499	10,199	11,899	13,599	15,298
Net lunar payload	1,911	3,358	4,816	6,266	7,719	9,175	10,622	12,073

Table 9. Minimum one-man mission concept

Step	Spacecraft	Booster	Mission
1	*Ranger* and/or *Surveyor*	A-A	Select landing site and gather knowledge on the site.
2	*Surveyor*	A-C	Establish radio "homing" beacon on-site. Establish visual monitoring capability on-site.
3	Bus + propulsion payload	C-3	Cache return propulsion on lunar surface (minimum of 2 firings).
4	Bus + command module (unmanned) + E.C.S. package + vernier propellant unit	C-3	Cache return S/C for man on lunar surface. Cache vernier propellant for return transit maneuver into entry corridor.
5	– – – – – – –	– – –	Automatic refueling of return S/C (Step 4) with propulsion modules landed (Step 3). Complete system checkout.
6	Bus + command module (manned) + E.C.S. package + mission module	C-3	Carry man to lunar cache.
7	– – – – – – –	– – –	Checkout of return S/C established in Step 5 in preparation for return to Earth. (Note: In case the system established in Step 5 is not O.K., the man may replace selected subsystems or have return propulsion transferred to his own landed S/C for return.)
8	Bus + command module (manned) + E.C.S. package + extra vernier propellant unit	Lunar-assembled return system	Lunar launch, return transit and re-entry and recovery of man.

Table 10. Over-all weight summary for minimum one-man mission

Spacecraft			Weight, lb									
Description	Weight, lb		At Liftoff	At Injection	Prior to Main Retro	Completion of Main Retro	Touch-down	Lunar Payload	Prior to Lunar Takeoff	At Injection from Moon	In Near-Earth Vicinity	At Re-entry
	Item	Total										
1. Propulsion S/C			30,000	30,000	29,774	10,966	10,364	7,884	....	....	....	....
Bus	2,476											
Propulsion payload[a]	7,681											
Weight contingency	207	10,364										
(2 S/C required, no abort propulsion required)												
2. Command module S/C, unmanned			30,000	30,000	29,774	10,966	10,364	7,884	....	....	....	....
Bus	2,476											
Command module[b]	3,304											
E.S.C. package	330											
Vernier propulsion unit[c]	1,838											
Weight contingency	2,416	10,364										
(1 required, no abort propulsion required)												
3. Command module S/C, manned[d]			30,000	29,750	29,524	10,716	10,114	7,638	....	....	....	....
Bus	2,476											
Command module	3,484											
E.S.C. package	330											
Mission module	3,824	10,114										
(1 required, launching abort propulsion required)												
4. Return S/C, manned			....	....	....	....	....	....	19,848	7,204	6,800	3,484
Bus	2,476											
Command module	3,484											
E.S.C. package	330											
Additional vernier propulsion (for maneuvering to re-entry corridor) (1 required)	510	6,800										

[a]Replacement propulsion weight is 6,186 lb.

[b]No crew results in 180-lb weight reduction.

[c]Excess vernier propulsion to be used for return maneuver to desired re-entry corridor.

[d]Abort propulsion-weight penalty at injection = 250 lb.

Table 11. Weight summary for minimum one-man command module (capsule)

Item	Weight, lb		Remarks
	Item	Total	
1. *Structure*			All items are based on a review of the final *Apollo* study reports.
Heat shield	1155		
Airframe	548		
Aerodynamic surfaces (flaps)	250	1953	
2. Communications			Communications associated with manned system and those telecommunications for near-Earth usage (boost, re-entry).
VHF system (2 complete systems; transmitters, receiver, antennas, etc.)	20		
Data recorder	25		
PCM multiplexer	38	83	
3. Guidance and control			Only those guidance and control elements associated with re-entry are carried in the command module.
Aerodynamic flight control	100		
Reaction controls	40		
Radar altimeter	50		
Miscellaneous	25	215	
4. Power system			Power supply for boost and re-entry phases only.
Batteries and power regulator	75	75	
5. Cabling and connectors	35	35	
6. Life and life-support			
Crew	180		
Environmental control system	113		
Furnishings and equipment	280		
Instruments and displays	100	673	
7. Search and recovery			
Landing equipment (parachutes, shock absorber, flotation, etc.)	375		
Search and recovery aids (beacon, Sofar bomb, *radar chaff*)	75	450	
Minimum one-man capsule		3484	

Table 12. Weight summary for minimum one-man mission environmental control support package

Item	Weight, lb		Remarks
	Item	Total	
1. Auxiliary environmental control equipment			Based on final *Apollo* study reports.
Heat control	50		
Air circulation	30		
Humidity control	5		
O_2 and N_2 control	60		
CO_2 control	30		
Gas charge	100	275	
2. Structure and packaging	55	55	
Package		330	

Table 13. Weight summary for minimum one-man mission module

Item	Weight, lb		Remarks
	Item	Total	
1. Structure (module structure and hatches, etc.)	1000	1000	Based on final *Apollo* study reports.
2. Scientific equipment	200	200	
3. Environmental control (interface equipment, ducts, wiring, etc.)	50	50	
4. Furnishings and equipment	40	40	
5. Scientific laboratory to be emplaced on the Moon (instruments and required structure and support)	2534	2534	
Mission module		3824	

rules, including the use of the C-3 launching vehicle. An attempt is made here to develop a minimum concept in terms of payload requirement which can still satisfy the broad aims of the initial lunar landing effort. Weight summaries for the minimum one-man mission are given in Tables 2 and 4 through 14.

The system elements for the one-man lunar-landing mission are as follows:

1. Command module
2. Spacecraft bus module
3. Environmental control support capsule
4. Mission module
5. Propulsion module

The functional aspects of these elements are, in general, the same as for the three-man mission. The command module weight is scaled in such a way that the total habitable volume for the single occupant is approximately equal to that assigned to each man in the more ambitious mission (about 340 ft^3 per man, including volume occupied by internal equipment). In this case the command module and the bus form the basic manned transit unit both to and from the Moon.

With the exception of some minor adjustments, the weights presented for the command module are the same as those shown in JPL's contribution to the NASA Headquarters study report, "A Survey of Various Vehicle Systems for the Manned Lunar Landing Mission." The

Table 14. Weight summary for minimum one-man mission propulsion payload (2 required)

Item	Weight, lb		Remarks
	Item	Total	
1. Propulsion module for refueling			Two modules required to refuel a landing bus/capsule S/C.
Main booster			
Inert weight	248		
Propellants	5938	6186	
2. Refueling equipment			
Propulsion-transfer equipment (hoist, track, actuators, etc.)	1125		
Monitoring equipment (TV monitor)	30	1155	Utilizes bus power and rf link.
3. Environmental control			
Container, etc.	300		
Monitoring T/M	40	340	Utilizes bus power and rf link.
Propulsion payload		7681	

weights as presented in that study report were derived from interim reports of the *Apollo* Industry Study and were interpreted to represent a "full-scale" *Apollo* command module. However, a review of the final reports on the studies has shown that the weights are somewhat optimistic and are, in fact, directly applicable to the somewhat scaled-down module indicated in the present study (about 0.8 the volume of the three-man *Apollo* size).

F. Typical Operational Sequence

In order to ensure the early accomplishment of the manned lunar mission, maximum utilization must be made of the techniques and operations which have been developed on preceding lunar missions. The operational sequence presented here (Table 15) makes use of the spacecraft technology which has been or is being evolved from the *Ranger*, *Surveyor*, and *Prospector* missions. The requirement to place several spacecraft at a given site on the lunar surface can be considered as an extension of the *Surveyor* guidance and propulsion systems.

For maximum reliability, the functioning of the various spacecraft subsystems will be established prior to use on the manned mission.

In addition to maximum utilization of the preceding lunar spacecraft technology, it is planned that *Surveyors* whose payloads are modified for the purpose will be used in direct support of the manned mission.

The operational sequence should be considered typical of one which would describe the mission. Variations according to the several payloads which the bus accomodates are indicated in the vernier (terminal) phase of the lunar descent.

G. Landing Guidance Accuracy

Two guidance-accuracy figures are of interest for lunar missions: (1) the proximity with which a payload can be delivered to a given point on the surface; (2) the accuracy with which it can be *determined* where the payload has been delivered. The accuracy required is strongly dependent upon the mission to be performed.

For the *Ranger* missions, it is sufficient to land within 100 km (3σ) of a nominal impact location. The landing point must be known, however, to within 35 km because the payload retro-motor must be aligned within 1 deg of the approach velocity vector. The first requirement is met by performing a single radio-commanded trajectory-correction maneuver about 16 hr after injection. The orbit determination is based upon angle and range-rate (doppler) data from the DSIF stations.

The early *Surveyor* missions will require similar accuracy. It is expected that landing accuracy of 75 km (3σ) will be achieved, assuming that a single midcourse correction is performed as described above. A second midcourse correction made approximately 24 hr after the first correction could further reduce the error in landing; however, greater precision in orbit determination would be required to make a significant correction.

It is desirable that later *Surveyor* payloads include ranging equipment. The use of range data for trajectory calculations would permit a second guidance correction to achieve 10 km accuracy. Figure 8 shows the reduction in impact error with successive guidance corrections, assuming ranging is used.

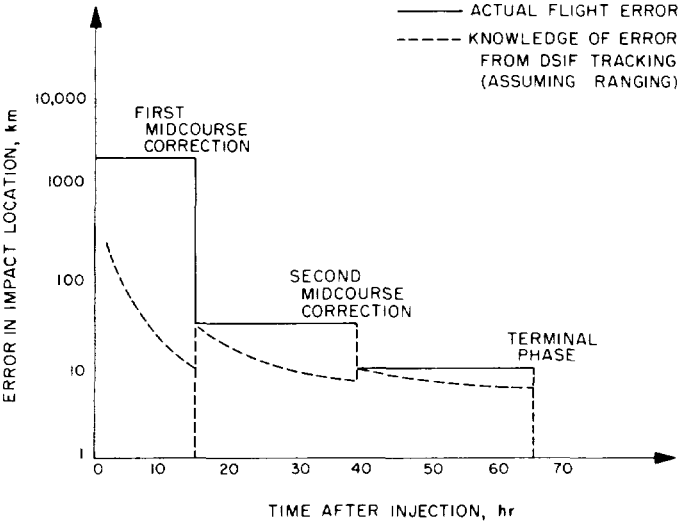

Figure 8. Typical three-sigma errors in impact location vs time-after-injection

For the lunar surface rendezvous missions, it appears desirable to deliver a payload to within a few meters (or less) of a specified point on the lunar surface relative to an already existing payload or equipment. Guidance to this accuracy would require "homing" techniques to remove the nominal 10-km landing dispersion.

For a *Surveyor*-type landing, it appears most efficient to initiate "homing" when the vernier descent is begun, following main retro burnout (at about 8-km altitude). During a normal vernier descent, the spacecraft is under the control of variable-thrust engines which are throttled to provide the proper deceleration rates. Both vertical and horizontal velocity are sensed with a doppler radar.

Table 15. Typical flight operational sequence

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
1	Lift-off	(4) Caged.			(3) Transfer to internal auxiliary power.	(2) Voice link, telemeter functions.	(1) Maximum restraint.	Assume crew monitors and report all functions.
2	Final boost to injection	(1) Uncaged. (2) Stabilization of final stage shutoff transients.	(3) Initiation of S/C programmer.					
3	Coasting flight (pre-midcourse maneuver)	(3) Reduction of rates about roll, pitch, yaw, axes. (4) Roll axis pointed toward Sun and star seeker unshuttered. (5) Star acquisition by programmed roll about S/C Sun axis.	(8) S/C trajectory determination by ground tracking to achieve nominal trajectory to arrive at lunar rendezvous site.		(2) Unfold solar array. Switch from auxiliary to main power-battery recharge.	(6) High-gain antenna pointed to Earth by ground command.	(7) Subsystem monitoring. Scientific observations, video observations, rest.	(1) Separation of S/C from final stage and deployment of legs.
4	First midcourse maneuver	(4) Align S/C roll and pitch axis.	(3) Store midcourse commands. (6) Initiate thrust signal. (8) Thrust cutoff signal from accelerometer.	(7) Thrust on. (9) Thrust off.	(5) Switch to auxiliary power.	(1) Ground command to initiate guidance computer, align roll and pitch attitude for correction. (2) Ground command for magnitude of ΔV.		

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
5	Coasting flight (post-midcourse maneuver) (pre-terminal maneuver)	(1) Roll axis pointed toward Sun and star seeker unshuttered. (2) Star acquisition by programmed roll about S/C Sun axis.			(5) Switch from auxiliary to main power.	(3) High-gain antenna pointed to Earth. (4) Tracking in preparation for terminal maneuvers.	(6) Rest and observation.	
6	Second midcourse maneuver	Operations (2) through (8) are identical to sequence in first midcourse maneuver.				(1) Ground tracking using DSIF ranging to establish magnitude and direction of ΔV.		
7	Preparation in terminal maneuver	(3) Execute maneuver for main engine attitude [step (1)-(a)]. Orient to velocity vector.	(2) Store ground commands of (1)-(a) through (1)-(f).		(6) Switch to auxiliary power.	(1) Ground commands for storage for terminal maneuver. (a) Roll, pitch, yaw for main engine attitude. (b) Pitch and yaw commands for realignment after main engine burnout.		(4) Deployment of high-altitude altimeter antenna. (7) Retraction of any panels, arms in preparation of retro thrust.

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
(No. 7 cont'd)						(c) Azimuth and elevation commands for positioning of high-gain antenna during main engine burning. (d) Azimuth and elevation commands for positioning high-gain antenna at start of vernier phase. (e) Timer setting for engine initiation sequence. (f) High-gain antenna steering matrix coefficients for vernier phase. (5) High-gain antenna positioned toward Earth [step (1)-(c)].	(8) Subsystem monitoring, operational checkout of all functions, check and compare with ground for decision making on terminal maneuver go-ahead. (9) Maximum restraint assumed for either go-ahead or turn-around.	
8	Main retro maneuver	(4) Control attitude to maintain roll axis on velocity vector. (8) Re-align roll axis to local vertical.	(2) Initiate guidance to maintain attitude on velocity vector. (5) Initiate main retro engine.	(3) Initiate vernier engine. (6) Main retro engine on. (7) Main retro engine burnout.		(1) Altimeter signal. (9) Reposition high-gain antenna to Earth.	(11) Maximum restraint.	(10) Jettison main retro engine module after burnout.

Note: Numbers in parentheses indicate sequence of operations for that phase.

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Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
9 9A	Vernier phases for various payloads Site-selection payload vernier phase (*Surveyor* with hover and translational primary capability and TV reconnaissance)	(8) Maintain roll axis of S/C to lunar vertical during all hover operations.	(4) Initiate closed-loop guidance for descent to predetermined altitude. (6) Initiate hover and translation mode. (13) Selected-site approach.	(1) Vernier engine maintained on. (7) Modulate and vector vernier thrust to hover and translate. (14) Modulate vernier thrust to descend. (16) Vernier letdown and shut-off.		(2) Doppler radar acquisition. (3) Altimeter on. (5) Altimeter signal to commence hover mode. (9) High-gain antenna pointing to Earth. (10) Monitor wide-angle TV and command translation maneuver to investigate favorable sites. (11) Command descent and hover for detailed viewing of selected site. (12) Monitor reserve hover and translation capability. (15) Detailed examination performed. Command final descent.		

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
9B	Refueling payload vernier phase (MOM bus + propellant payload)	(7) Maintain S/C roll axis to lunar vertical during hovering and final descent. (10) Roll S/C axis to proper orientation.	(4) Initiate closed-loop guidance with lunar-based beacon homing. (5) Change from coarse homing to vernier homing on final approach and hover. (9) Go from hover mode into final descent.	(1) Vernier engine maintained on. (5) Vector vernier thrust for proper translation during decent. (11) Final vernier engine letdown and shut-off.		(2) Doppler radar and rendezvous site beacon acquisition. (3) Altimeter on. (6) TV monitor rendezvous site during descent phase. (8) Survey site prior to final descent. Command S/C orientation with respect to landed payloads.		
9C	Return S/C (MOM bus + return S/C	**NOTE** Vernier phase for this S/C identical to 8B and orientation made with respect to previously landed refueling payloads. Landing and orientation made by combination of TV monitoring with Earth commands and the lunar-based beacons.						
9D	Manned S/C (MOM bus + manned command module)	**NOTE** Vernier phase for this payload identical to 8C with the exception that S/C does not need to be located precisely with respect to other payloads in the lunar-rendezvous site. Distance from return S/C depends upon distance man can conveniently traverse.						
10	Touchdown				(3) Transfer to main power.	(4) High-gain antenna (tracking) pointing to Earth.		(1) Level and anchor S/C. (2) Deploy panels and any mechanical arms, scientific instruments, etc.

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
11	Lunar operations based on staging (rendezvous) on Moon	**NOTE** Operations (3), (4), (5), (6), (7) and (8) are performed with bus and propulsion payloads and bus and unmanned capsule.	(10) Obtain fix on landing site from Earth-Sun-star reference. (11) Realign platform and make settings for return.	(9) Monitor status of lift-off system.		(2) Transmit visual data from site to Earth. (3) Command Moon-based lift-off rocket or fuel module to rendezvous with S/C.	(1) Perform visual scan of landing site. Selected areas studied. (4) Monitor location and progress of rendezvous of return supply with S/C. (6) Checkout refueling or transfer prior to operation. (7) Initiate automatic transfer or hook-up or perform outside S/C. (4)-(a) Crew goes outside of S/C and performs mechanical functions required for refueling. (4)-(b) Perform scientific experiments and measurements on surface.	(5) Perform connections for refueling or transfer of lift-off units to S/C. (8) Commence transfer of fuel or units to S/C.
12	Lift-off-lunar phase	(7) Uncaged, monitor programmed attitude.	(6) Initiation of return S/C programmer, autopilot.	(9) Initiate main lift-off thrust. (10) Main thrust burnout.	(8) Transfer to auxiliary power.	(2) Transmit return commands and data. (3) High-gain antenna folded.	(1) Maximum restraint.	(4) Disengage return S/C from landing gear and equipment. (5) Align return S/C to proper azimuth and elevation.

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
13	Coasting phase (pre-midcourse)	(1) Stabilize thrust burnout transients and establish roll axis pointed to Sun and unshutter star-seeker. Star acquisition by programmed roll about S/C-Sun axis.	(4) Commence establishing ephemeris for return trajectory.		(5) Transfer to main S/C power.	(3) High-gain antenna pointed to Earth by ground command. (6) Transmit commands for storage in S/C for midcourse ΔV and time for ΔV.	(7) Release from maximum restraint monitor and report. Rest.	(2) Extend panels and high-gain antenna.
14	Midcourse maneuver	Repeat basic operations as in 4.						
15	Coasting phase (post-midcourse)	Repeat basic operations as in 5.				(1) Establish Earth-impact prediction and commence initiation of Earth-air-sea rescue sequence.		
16	Re-entry maneuver	(5) Stabilize to attitude for re-entry.	(9) Store commands for re-entry of man. (10) Align platform and set-up guidance computer conditions.	(7) Activate and ready reaction jet system.	(2) Transfer to auxiliary power.	(3) Commence tracking by *Mercury* net. (8) Transmit commands for re-entry trajectory.	(4) Monitor re-entry staging; perform steps as required. Transfer to maximum restraint.	(1) Begin re-entry staging sequence: (a) Release solar array. (b) Radiators. (c) Fairings. (d) On-board propellants. (e) Supporting structures, etc. (6) Deploy re-entry stabilizers and drag units.

Note: Numbers in parentheses indicate sequence of operations for that phase.

Table 15 (Cont'd)

No.	Mission Phase (Flight Sequence)	OPERATION						
		Attitude Control	Guidance	Propulsion	Power	Telemetry and Communications	Crew	Mechanical Functions and Miscellaneous Operations
17	Terminal maneuver (Atmospheric re-entry)	(4) Maintain the re-entry attitude with correct angle attack. (5) Transition from reaction to aerodynamic controls.	(1) Commence computer sequence for impact prediction. Store commands for lift/drag maneuvers to maintain re-entry corridor conditions. (3) Transmit attitude commands to attitude control system and continue computation on impact prediction.	(2) Reaction jet system active.		(6) Continue ground tracking, radar, optical. Backup computation of re-entry maneuvers. (9) Transmit position information to S/C prior to rf blackout.	(7) Change from maximum restraint to descent mode. (8) Monitor impact-point prediction computer and transmit information to ground-tracking net.	
18	Terminal maneuver (Ranging mode and recovery)	(2) Aerodynamic controls active.	(1) Compute range and cross-range data. (5) Initiate chute-deployment command at suitable altitude by baroswitch.			(4) Continue ground tracking. (9) Track and locate recovery beacons. (10) Direct recovery units to landing site.	(3) Perform computed maneuvers as required to impact selected target.	(6) Deploy drag chute. (7) Deploy ribbon chute at lower Mach No. and altitude. (8) Recovery sequence activated on touchdown.

Note: Numbers in parentheses indicate sequence of operations for that phase.

The spacecraft is tipped off-the-vertical in order to remove horizontal velocity which exists following main retro burnout. When "homing" is performed, the same technique is used to translate toward the lunar-based beacon.

A variety of equipment is available to determine range and bearing to a previously landed beacon. Commercial airliners use direction-finding and distance-measuring devices to accuracies of a few degrees and on the order of a kilometer. The use of a transponder will provide range to 50 meters.

If greater accuracy is required, other techniques can be used. Precise "homing" can be performed with various electronic and optical devices to centimeter accuracy. It is also reasonable to consider Earth-based television monitoring and control during the final hover and descent phase.

In order to provide sufficient hovering capability to remove a 10-km landing error, approximately 8% of the landed payload would be required for additional vernier propellant, assuming an I_{sp} of 300 sec. For a *Surveyor*, this would correspond to about 60 lb of payload, and the requirement varies directly with the landed weight. The total weight of electronic gear required on-board the spacecraft is less than 40 lb and is essentially independent of the size of the landed vehicle.

A number of methods are possible to reduce the vernier-propellant penalty described above. For example, timing of the terminal sequence could be changed in order to optimize the utilization of vernier propellant. This would involve ignition at a somewhat higher altitude and the removal of a smaller velocity increment with the main retro motor and a larger increment with the vernier engines. A systems analysis would have to be performed in order to determine the optimum parameters, but certainly a significant part of the vernier-propulsion penalty would be eliminated. Figure 9 shows the reduction in landing error which is achieved with both of these methods.

In conclusion, it appears that "homing" to the required precision is easily accomplished. The fuel and weight allowances described are extremely conservative and, yet, they are small enough that the "homing" technique can be readily demonstrated with a *Surveyor* spacecraft.

H. Lunar Refueling Operations

The refueling operation can be mechanized in one of several ways. A great amount of technology as well as many off-the-shelf components exist today which are applicable to this type of operation. In general, the operation involves transferring propellant from the refueling spacecraft to the return spacecraft, which can be accomplished by effecting the transfer either above the surface or on the surface.

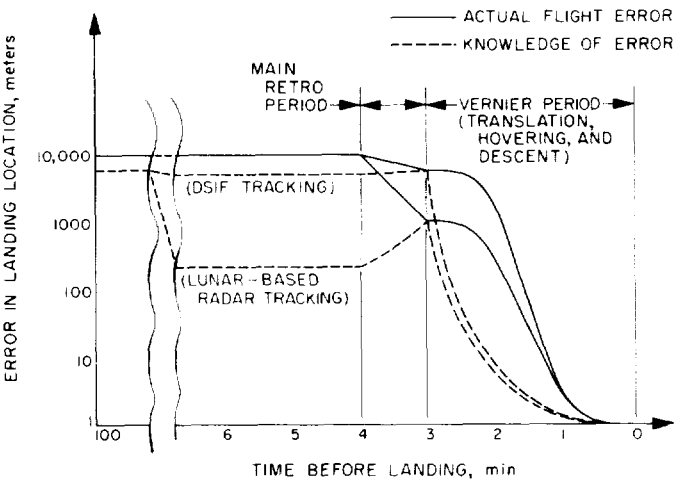

Figure 9. Typical three-sigma errors in landing vs time-before-landing

Whether liquid or solid propellants are used, similar techniques are applicable. However, solid propellants are being considered for application at this time.

1. Above-the-Surface Transport

One of a number of suitable above-the-surface systems is described to illustrate that remote operations, such as assembling or refueling a return spacecraft on the lunar surface, are feasible (see Tables 16 and 17). The operation involves transferring a solid-propellant rocket from the refueling spacecraft to the return spacecraft. This type of transport would involve mechanisms such as an overhead extendable boom, or a cableway, along which a trolley travels, carrying the rocket motor. The return spacecraft would be located at a maximum distance of 42 ft from the refueling spacecraft and within an azimuth angle of approximately 30 deg.

a. Transport mechanization. Figure 10 shows a general arrangement of the return spacecraft with respect to the refueling spacecraft in which the refueling vehicles are spaced about the centrally located return spacecraft. The transfer mechanism would consist of a carriage which rotates in azimuth, an extendable boom which is adjustable in elevation, and a trolley which transports the rocket motor along the boom to the return spacecraft.

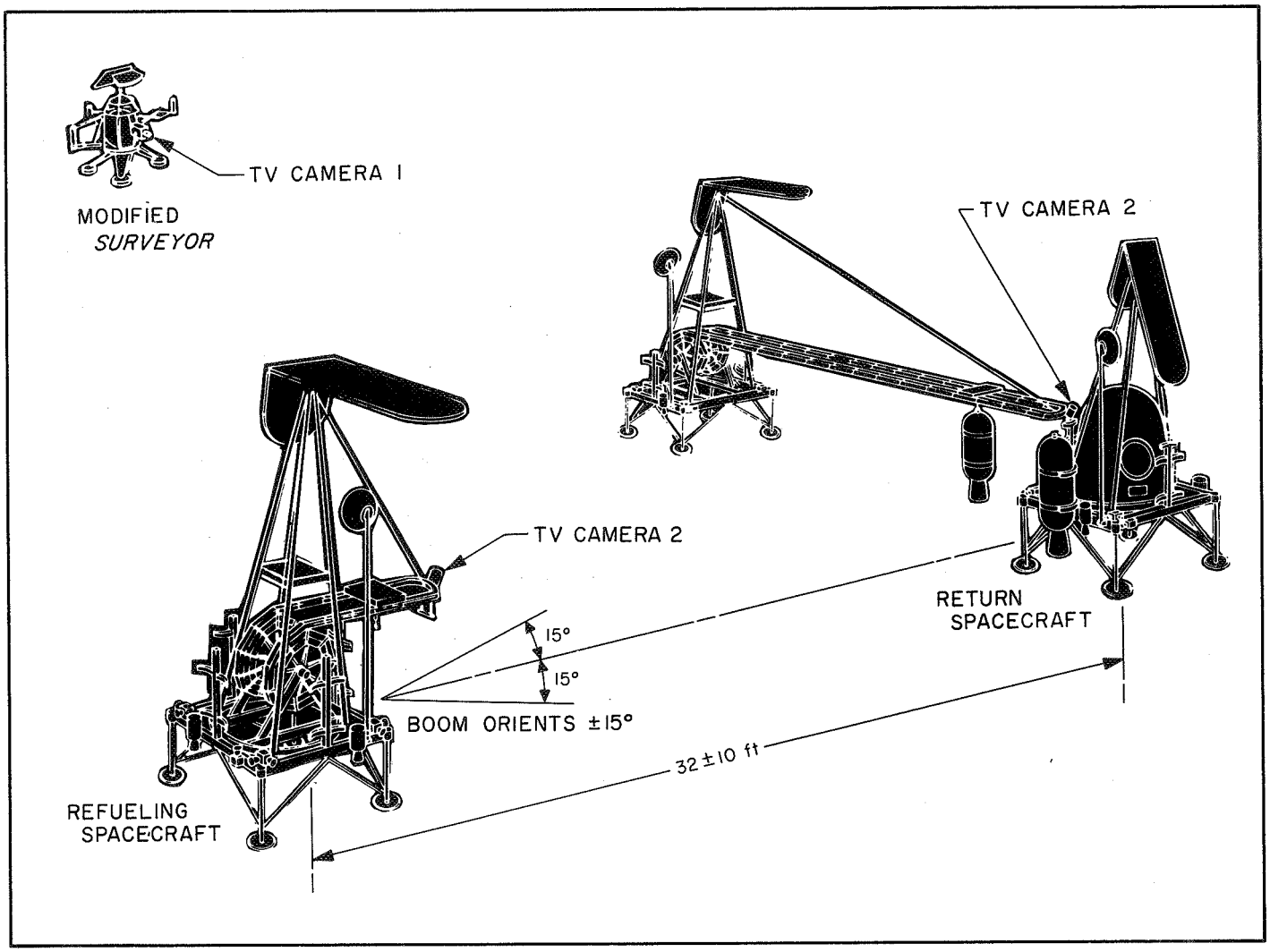

Figure 10. Typical above-the-surface lunar operations

The rocket-motor mount on the return spacecraft would grasp the rocket and automatically align the thrust axis. Connection of the igniter to the igniter circuit, as well as checkout of the igniter circuit, would also be automatic. The trolley would be disconnected from the rocket motor, and it and the boom would return to their original positions.

b. TV systems. A previously placed modified *Surveyor* spacecraft with a TV camera would provide surveillance of the terrain as well as monitoring of landing of the return and the refueling spacecraft. In addition, it would monitor various refueling spacecraft functions, such as carriage orientation, boom extension, trolley travel, et cetera. An additional TV camera on each of the booms would serve to align the boom with respect to the vehicle as well as monitor boom extension and rocket-motor emplacement.

c. Commands. It is estimated that approximately thirty commands would be required to orient and extend the boom and to refuel the return spacecraft. For reasons of simplicity, there would be no proportional commands; all commands would be of the on-off type.

d. Telemetry. In the interest of simplification, the majority of the operations associated with refueling would be monitored by TV. Certain operations such as latching and unlatching, which cannot be easily monitored with TV, would be monitored by telemetry using appropriate transducers and transmitting instrumentation.

Table 16. Typical above-the-surface lunar operations sequence

Operation		T/M Required	Remarks
No.	Description		
1	Turn on TV camera 1 to obtain a general view of the return and refueling spacecraft.	No	This camera shall have a resolution of approximately 1.0 in. at 1000 ft.
2	Turn on TV camera 2. Camera axis to be parallel to the boom axis. (Camera is used for aligning boom with respect to return spacecraft.)	No	Camera shall have a resolution of 0.3 in. at 30 ft, a minimum field of view of 30 deg and be capable of being focused from 60 ft to 2 ft.
3	Rotate carriage to align boom axis in azimuth with respect to refueling spacecraft.	No	Commands are to be on-off commands only. There will be no proportional commands. Carriage supports and rotates boom approximately 30 deg in azimuth.
4	Unlatch boom.	Yes	Boom is a self-supporting cantilever during extension. At completion of extension, the outboard end attaches to the return spacecraft and it becomes a simple beam capable of supporting the solid-propellant motor. Maximum extension is 40 ft.
5	Align boom axis in elevation using boom-mounted TV camera 2. Cross hairs in camera lens align with target on return spacecraft.	No	Carriage and boom elevation drives receive power from the central source on the bus, which may be a combination of batteries and solar panels for longevity.
6	Start boom extension movement and monitor with TV cameras.	No	Boom continues to extend until it enters self-aligning socket on return spacecraft, at which time it is stopped by limit switch.
7	Release rocket motor from stowage cradle.	Yes	Release mechanism is automatically operated.
8	Start trolley. Carry rocket motor to return spacecraft.	No	Drive mechanism shall be mounted on trolley.
9	Stop trolley.	Yes	Limit switch on trolley stops trolley drive when rocket motor has reached position.
10	Rocket-motor mount on return spacecraft receives rocket.	Yes	Clamping device can receive rocket from 15 deg either side of center. Motor drive on clamp.
11	Check continuity of igniter circuit.	Yes	Continuity check instrumentation is on refueling spacecraft.
12	Disengage trolley from rocket motor.	Yes	Same as Operation No. 7.
13	Start trolley return.	No	TV monitor.
14	Stop trolley return.	No	
15	Start boom return.	No	
16	Stop boom return.	No	
17	Perform systems test on return spacecraft.	Yes	Turn on spacecraft power and verify that all spacecraft elements will function properly and that spacecraft is ready for takeoff.

*e. **Advantages.*** This system, by bridging such obstacles as crevices and boulders, would perform the transport function independently of local disturbances and discontinuities in the lunar surface between the refueling spacecraft and the return spacecraft. In addition, the use of this system would not require equipment other than that already being carried on the refueling payload.

*f. **Disadvantages.*** The landing accuracy of each unit would have to be contained within fairly narrow limits because of the fixed distance that a boom or cableway can be extended. Also, it is a single-purpose unit with limited usefulness for other lunar manipulating operations.

2. On-the-Surface Transport

On-the-surface transport would require the use of a roving transport vehicle with a mechanical arm or similar manipulating device. The operation would involve the use of several refueling payloads, the transport vehicle, and the return spacecraft vehicle. In the discussion which follows, a possible approach to assembly by means of a roving vehicle is given. Many other types of mechanization are possible.

The refueling payloads could be located at any distance from the return spacecraft. The distance would be limited

Table 17. Typical on-the-surface lunar operations sequence

Operation		T/M Required	Remarks
No.	Description		
1	Turn on TV camera 1 to obtain a general view of the operational area and the refueling, return, and transport payloads.	No	This camera is mounted on a previously landed *Surveyor* and has a resolution of approximately 1.0 in. at 100 ft.
2	Unlatch transporter from bus.	Yes	The bus is identical to the one carrying the refueling rocket and the capsule.
3	Select and lower appropriate ramp.	No	Two ramps will be provided so as to permit selection of exit from the bus should one of the exits be obstructed.
4	Turn on TV camera 2. Camera is mounted on the manipulator arm and is used to monitor the transfer of the rocket as well as to view the terrain in the direction of transporter travel.	No	Camera shall have a resolution of 0.3 in. at 30 ft and shall be capable of being focused from 3 ft to ∞.
5	Command transporter down ramp.	No	Transporter is a roving vehicle with self-contained power source and capable of being maneuvered either by Earth command or a homing control.
6	Using TV camera 1 to monitor, steer the transporter in the direction of the refueling spacecraft.	No	Steering is by Earth command through an omni-antenna.
7	Using both TV cameras, steer transporter around boulders, crevices, and other surface discontinuities.	No	
8	Turn on guidance beacon on refueling spacecraft.	Yes	Beacon may be either optical or rf, depending on a later decision. This operational sequence will be based on an rf beacon.
9	Erect directional antenna on transporter.	No	TV is used to steer the transporter to a point within the beacon beam.
10	Switch transporter control from Earth command to homing.	No	Switching is done when it is certain that no further obstructions exist between the transporter and the refueling spacecraft.
11	Transporter stops when engagement has been made with refueling spacecraft.	No	Limit switch is actuated when transporter is in proper position.
12	Lower manipulator arm and engage handling lugs on rocket. TV camera 2 monitors these movements.	No	Action is controlled by Earth command. Attachment device is guided by self-aligning socket on manipulator arm.
13	Unlatch rocket motor from bus.	Yes	Actuated by above limit switch.
14	Hoist rocket motor and transfer to the transporter.	No	Rocket is carried on transporter so that center of gravity is not raised. Actuated by time-delay relay.
15	Switch control from homing back to Earth command.	Yes	Earth command is used until the transporter is in the immediate vicinity of the return spacecraft.
16	Back transporter away from refueling spacecraft.	No	Earth command with TV surveillance.
17	Steer transporter in direction of return spacecraft.	No	A course is selected based on TV surveillance of the area between the refueling spacecraft and the return spacecraft.
18	Switch to homing control.	No	This is done as soon as it has been determined that a clear path exists ahead of the transporter.
19	Transporter stops when engagement has been made with return spacecraft.	No	Limit switch stops transporter drive when it has reached proper position relative to return spacecraft.
20	Transfer rocket motor to return spacecraft and lower into position.	No	This action is initiated by the limit switch referred to in Operation No. 19.
21	Rocket-motor mount on return spacecraft receives rocket and clamps it in place.	Yes	The clamping action is motor driven and automatically aligns the thrust axis.
22	Check igniter circuit continuity.	Yes	Instrumentation for circuit check is in the return spacecraft.
23	Back transporter away from return spacecraft.	No	
24	Perform systems test on assembled return spacecraft.	Yes	All elements of the return spacecraft are checked to verify that the spacecraft is ready for takeoff.

by the roving vehicle's available power and the viewing range of the TV camera. Figure 11 shows the transport vehicle with respect to the return and refueling spacecraft.

a. Transport vehicle. The transport vehicle would be a remotely controlled, battery-operated vehicle, equipped with a manipulating arm similar to those developed by industry for the Atomic Energy Commission and for general industrial use. An automatic latching device mounted at the end of the arm would be positioned in the vicinity of the rocket motor when the transport vehicle has reached the refueling payload. A self-centering device having a conic form would guide the latching devices onto the coupler on the rocket motor. The rocket motor would be unlatched and lifted clear of the refueling spacecraft and transported over the surface to the return spacecraft; here, the rocket-motor mount would grasp the rocket and automatically align the thrust axis. Connection of the igniter to the igniter circuit, as well as checkout of the igniter circuit, would be automatic.

b. TV systems. Two TV systems will be used: one system would provide general surveillance of the terrain, as well as monitoring of the landing of the refueling, transport, and return payloads. In addition, this system would monitor the refueling operations and also view the area in the direction of transport vehicle travel in order to avoid obstructions and other surface discontinuities. This camera would be located on a previously placed modified *Surveyor* spacecraft.

An additional TV camera mounted on the arm would give close-up surveillance of the transport operations, such as rocket-motor acquisition and emplacement. It would also serve to monitor the direction of vehicle travel.

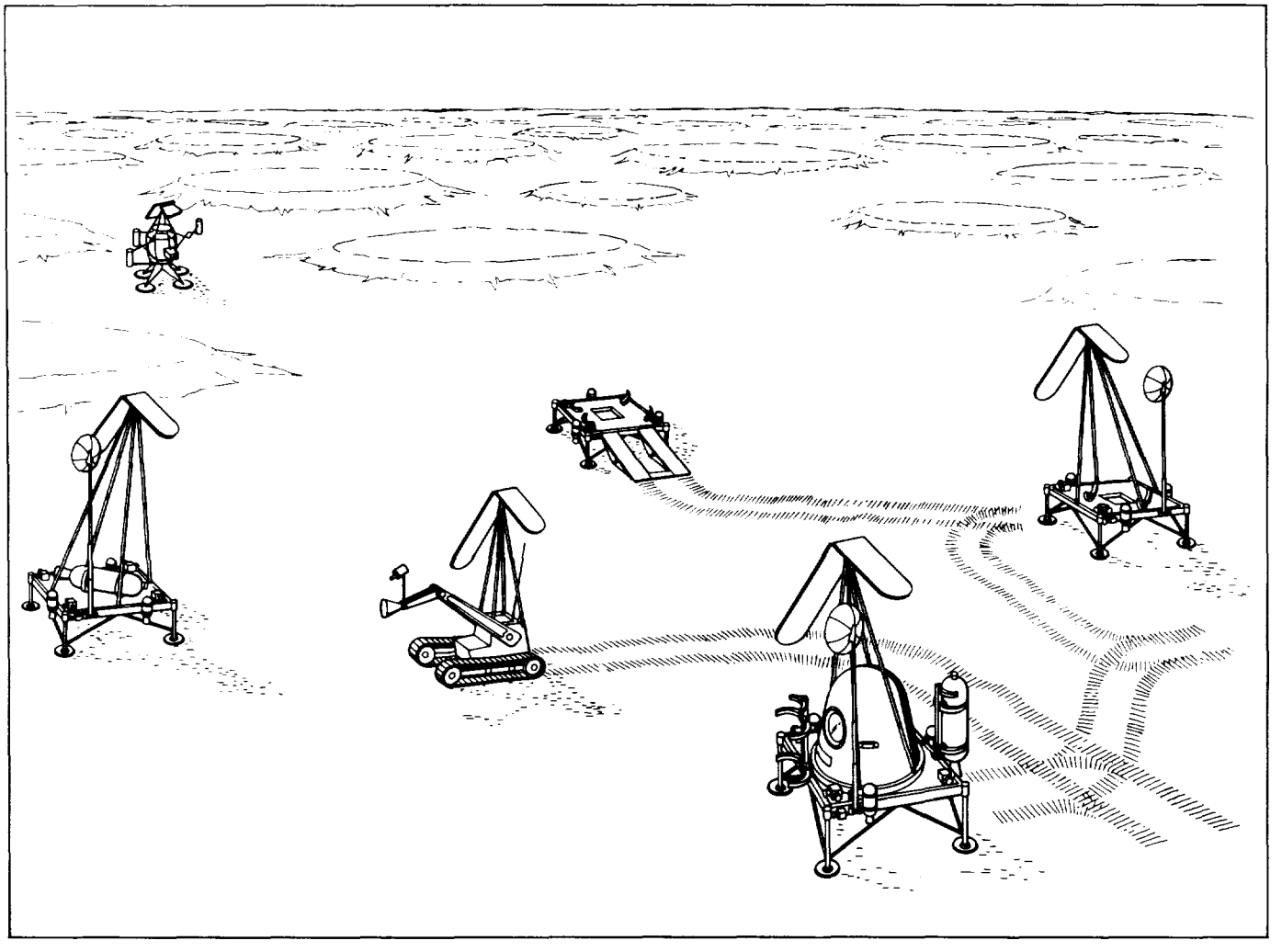

Figure 11. Typical on-the-surface lunar operations

c. Commands. Insofar as possible, Earth commands would be held to a minimum. Direction of travel would be controlled by the use of suitable beacons located on both the refueling and return spacecraft. Provision would be made for the use of over-riding Earth commands if, because of a malfunction in the control system, it should become necessary to control the vehicle by ground command. For simplicity, there would be no proportional Earth commands; all commands would be of the on-off type.

d. Telemetry. Telemetry would be similar in concept to that described for the "above-the-surface" transport mechanism.

e. Advantages. The required landing accuracy would have considerable latitude as the distance between the refueling payloads and return spacecraft would be limited only by the capacity of the roving vehicle to travel on the lunar surface and by the viewing range of the TV cameras. In addition, it could become a general-purpose unit which could perform such other lunar operations as remote assembly of structures and transportation of scientific mission packages.

f. Disadvantages. Certain lunar-surface obstructions and discontinuities, such as crevices and boulders, could impare its mobility. One additional payload would be required in addition to the refueling and return spacecraft payloads. This additional payload must function properly and maintain itself for the duration of the refueling operation.

I. Mission Schedule

A schedule for the manned lunar-landing and return mission by means of the lunar-surface rendezvous technique is shown in Fig. 12. The pacing item for mission accomplishment is launching vehicle development and launching rate. The major elements of the spacecraft system (manned capsule, spacecraft bus, and propulsion payload) require three to three and one-half years of design and development time, with the exact time being a function of the procedures followed in selecting contractors and awarding contracts.

The schedule shown in Fig. 12 is based on a *C*-3 availability date of July 1965, with a subsequent firing rate of two vehicles every ~~two~~ months (2 launch pads).

This schedule does not allow for a full unmanned system test (lunar assembly and return of the complete system). If such a full-scale test is desired, the number of firings required for the three-man mission increases to 15 and the direct program cost would increase proportionately.

Examination of the landing performance of the transit, landing, and return spacecraft indicates that it would be possible (by using the payload capability of this spacecraft to carry return propulsion) to return this spacecraft from the lunar surface to the near vicinity of Earth. Such a mission could demonstrate the lunar takeoff guidance, return midcourse guidance, and terminal maneuver guidance in preparation for the manned mission.

J. Cost and Funding Requirements

The total cost of the program for the three-man lunar-landing mission is 2.5 billion dollars. This amount includes the cost of the *Ranger* and *Surveyor* projects, launch vehicles, spacecraft, capsules, propellant, lunar complex, TV monitoring of the mission, and DSIF operations. The amount is based on a total of eight *C*-3 firings and assumes a 50% failure rate in the first six firings.

The total cost for the minimum one-man lunar-landing mission is 1.8 billion dollars. This amount includes the items listed above for the three-man mission and is based on a total of six *C*-3 firings with a 50% failure rate in the first four firings.

Cost breakdowns for the two missions are shown in Table 18.

The cost of the two *C*-3 launch facilities is not included.

Table 18. Funding requirements

Item	Flight Units	Cost X 10^{-6}								Σ
		CY	1961	1962	1963	1964	1965	1966	1967	
		FY	1961	1962	1963	1964	1965	1966	1967	
Ranger (spacecraft and vehicles)	9		20	85	50	7				162
Surveyor (spacecraft and vehicles)	18		2	40	120	120	60	18		360
Σ			22	125	170	127	60	18		522
Three-man lunar landing mission										
Vehicles										
C-3[a]	8			30	50	80	50	30		240
Little Joe II[b]	7			1	2	3	1			7
Atlas-Agena[b]	3			2	8	14				24
C-1[b]	17			30	60	80	60	25		255
Spacecraft[c]										
Bus	8			20	130	200	180	70		600
Manned payload	2			15	90	100	70	25		300
Propulsion payload and lunar operations	6			10	120	190	145	45		510
Flight operations	8			5	15	20	20	20		80
Σ for three-man mission				113	475	687	526	215		2016
Minimum one-man lunar landing mission										
Vehicles										
C-3[a]	6			20	40	60	40	20		180
Little Joe II[b]	7			1	2	3	1			7
Atlas-Agena[b]	3			2	8	14				24
C-1[b]	17			30	60	80	60	25		255
Spacecraft[c]										
Bus	6			20	100	150	140	40		450
Manned payload	2			15	60	90	60	25		250
Propulsion payload and lunar operations	4			10	80	130	90	30		340
Flight operations	6			5	10	15	15	15		60
Σ for one-man mission				103	360	542	406	155		1566

[a]Purchased; no development cost included.

[b]Vehicles for S/C development flights.

[c]Includes development and unit cost for complete S/C system.

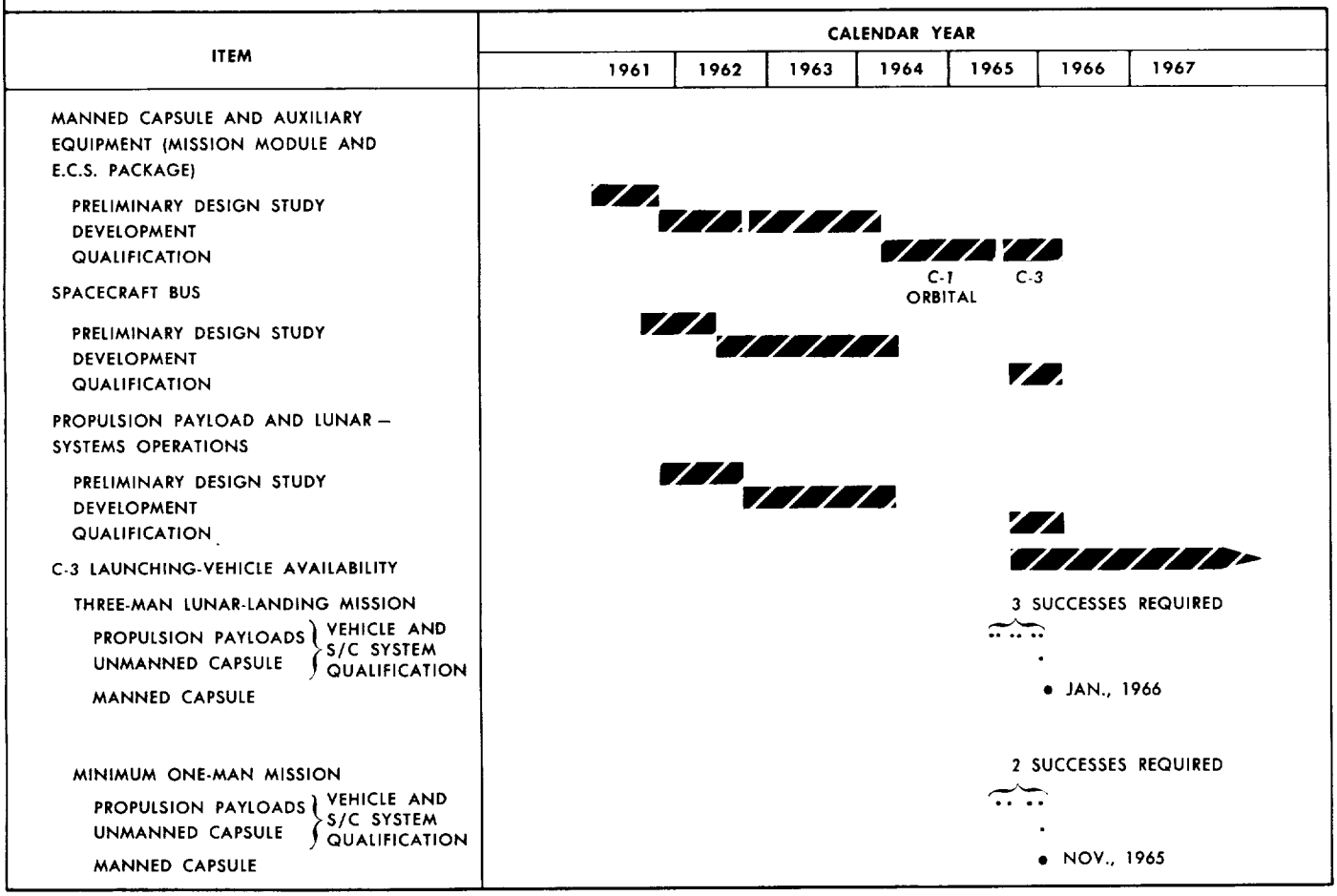

Figure 12. Schedule for lunar-surface rendezvous mission

II. A SOLID-PROPELLANT NOVA INJECTION VEHICLE SYSTEM

A. Introduction

With Man-on-the-Moon a major program objective, NASA now has under consideration large solid-propellant boosters as one potential launch method for accomplishing the *Nova* mission as a single operation. While the Jet Propulsion Laboratory is not assigned the responsibility for launching-vehicle development programs, it is felt that the Laboratory's extensive solid-propellant technology and vehicle system experience places it under obligation to submit any pertinent information that might expedite the manned lunar program or enhance the likelihood of its success.

Consequently, the Laboratory has undertaken a preliminary study[1] to answer several basic questions:

1. Is a solid-propellant *Nova* vehicle technically feasible?
2. If so, what might such a vehicle system consist of conceptually?
3. When can the vehicle be made available for a manned lunar mission?
4. What is the approximate cost of the program?
5. What are the major problem areas?

1. Background and Philosophy

The following assumption is considered fundamental:

Within the shortest possible time, NASA must send a man to the Moon and return him safely to Earth.

This may well imply that some type of parallel approach to putting a man on the Moon must be adopted in order to ensure early success. President Kennedy, in his recent address to Congress, did indeed call for both a liquid- and a solid-propellant booster approach for the *Nova* Moon mission. In such a costly parallel effort it appears judicious to avoid the use of any vehicle subsystem as a common element in both vehicles if there is a strong possibility that a delay in the program could arise from that common subsystem. If the liquid-propellant *Nova* schedule were to experience a delay, it might well arise from our limited liquid hydrogen technology and/or the cluster of 200K liquid hydrogen–liquid oxygen J-2 engines. One must conclude, therefore, that the solid-propellant *Nova* should, for now, use solid-propellant rockets throughout or, alternately, use solid-propellant rockets with relatively small, storable, liquid-propulsion systems which can be based on a more highly developed technology.

If the development of an all-solid-propellant vehicle is to be consistent with the above fundamental assumption, full advantage must be taken of our past experience and present knowledge. It should be recognized that development work on increasing the performance of chemical rocket systems is rapidly approaching a practical upper limit for large systems in which the development cost can be amortized over a very limited production run only. By minimizing a development program, both money and time can be saved. Therefore, a conservative vehicle design incorporating today's state-of-the-art has been adopted for this study.

For a given payload these constraints produce a larger vehicle, but it is believed that the larger size introduces relatively minor problems compared to those arising from a lighter-weight vehicle with a much more advanced design. In addition, it is believed that the more conservative vehicle provides, inherently, a man-rated system at a much earlier point in the flight program and at a reduced cost.

Past experience also indicates that the program should capitalize on rocket motor scaling principles. This powerful tool of scaling was demonstrated to be applicable to solid-propellant rockets in the *Sergeant* Program, when a flight-weight scale model of the nozzle, chamber, and propellant charge accurately predicted the performance and characteristics of the full-scale, flight-weight motor, a unit which weighed 125 times as much as the model. *Pershing*, *Polaris*, and *Minuteman* have extended in size the applicability of the principle. When this scaling ability is coupled with the decision to use state-of-the-art technology, the need for a static test program for the motor is reduced essentially to confirming, on the large-scale design, characteristics which have already been proven on a smaller scale, i.e., demonstrating that extrapolation of the scaling law extends to and includes the new, larger size.

Again, solid-propellant rocket experience should be helpful in that it has revealed the desirability of initiating flight tests of the vehicle system at the earliest possible

[1]Additional material in support of this study is contained in Technical Memorandum No. 33-52, Addendum A, Jet Propulsion Laboratory, Pasadena, Calif.

date. In solid-rocket propulsion, static and flight tests with "battleship" hardware tend to be program diversions of very limited value. Static tests on stages of clustered motors are unnecessary. (The high-speed clustered stages used in the *Explorer* and *Juno* Programs were developed without ground testing of clustered motors. The validity of this procedure was demonstrated by the successful flight program.)

The large size of the *Nova*-class vehicle allows for abnormally large weights and volumes for guidance, control, and auxiliary subsystems. This fact should remove one of the serious constraints imposed on current guidance and auxiliary systems and hence will allow greater freedom in the choices of proven elements, in the application of proven techniques, and in the judicious use of redundancy. The over-all vehicle system reliability should be significantly improved by capitalizing on these advantages.

Because of the above decisions and system characteristics, a program approach that is fundamentally different from that of past launching vehicle systems is advocated. The program philosophy would resemble that for bridge building more than for aircraft or missile construction. In the latter, standard approach for missile construction, testing of components and assemblies occurs after each phase in the development and before successive phases are begun. Components are tested, then combined as subassemblies for test; the subassemblies are then combined as the first stage for flight test; then the first and second stages are assembled for flight test, etc. In the new approach the vehicle is designed conservatively, analyzed using known principles, then assembled, like a bridge, for immediate use. Testing is performed primarily to confirm what is known. An important point to remember when applying this philosophy is that weight, *per se,* is not a technical obstacle. It should be considered limiting only if it grossly affects costs or technical feasibility. The engineering design philosophy recommended is one which trades size and weight for time and money.

It is important that individuals and/or organizations assigned to carry out portions of a major program such as discussed in this study participate in its early formulative stages. This approach not only stimulates their interest but provides an excellent mechanism for incorporating their thoughts and experience into the development of requirements to which they will later be committed. This philosophy is an important element in achieving early mission success.

It is assumed for the purpose of this study that the leading mission characteristics will compete in the following order:

1. Mission success (schedules and reliability).
2. Cost.
3. Growth potential.

It must be pointed out that the study has been concerned primarily with a vehicle capable of injecting 130,000 lb to escape velocity. This injection capacity was adopted arbitrarily as being representative of the current *Apollo* spacecraft studies and is here used to establish the launching vehicle scale.

The vehicle discussed is conceptual in nature and is not intended to represent a final design; rather it indicates a feasible system. Indeed, it should be stressed that further comprehensive study (as indicated in Section II-D) would be necessary if consideration were given to implementing a solid-propellant *Nova* program.

The *Apollo* spacecraft was examined briefly but only to obtain some indication of the vehicle spacecraft interface problems and for assurance that some characteristic of a solid-propellant vehicle would not prevent satisfactory operation of the spacecraft.

2. Summary

This study shows that an all-solid-propellant *Nova* injection vehicle system is technically feasible. The vehicle studied consists of a four-stage solid-propellant rocket having a gross weight in the 25,000,000-lb class. The first three steps will inject the fourth stage into a parking orbit from which the fourth step injects the spacecraft into a transfer orbit to the Moon.

From a technical standpoint, it is believed that the vehicle injection system can be made available for a manned lunar landing and return five years after the date of go-ahead. Total costs for a flight program of 20 vehicles are estimated at about $2.6 billion and include vehicle and spacecraft development and production costs, all special production facilities such as the propellant plant, launch facilities and GSE, and launch operations. The costs of astronaut training and the flight test program for spacecraft development prior to the *Nova* phase were omitted.

At this time there do not appear to be any major technical problem areas; however, early emphasis should be given to thrust vector control and to meeting guidance and control as well as other subsystem reliability re-

quirements. Combustion instability is not expected; it is believed that even if it is encountered it would not become a serious problem.

As a result of these studies the confidence of the study group in the philosophy and program approach advocated above has been reinforced. It is concluded that this approach has merits at least equal to those of any other launch vehicle system and full consideration must be given the solid-propellant *Nova* approach before final selection is made of the system or systems to be developed for putting a man on the Moon.

B. Mission Approach

The single most important question that must be answered before deciding on an over-all design philosophy relates to the acceptable level of risk associated with the mission. Many decisions of far-reaching consequence cannot be intelligently made until the level of risk is established. Some of these decisions relate to:

1. The level of reliability that must be designed into the system.
2. The amount and kind of abort capability that must be built into the system.
3. The amount of shielding that is required.

The problem of determining the actual level of system reliability is very difficult—or even impractical—to establish quantitatively. If reliance were to be placed on a straightforward flight test program to establish a demonstrated reliability record, very large numbers of test flights would be required. As an example, a series of ten successful operations with no failures implies, by elementary statistical theory, a 50% confidence level that the failure rate is less than 5% or a 90% confidence level that the failure rate is less than 25%. It thus does not appear likely that any program of the type under consideration will be based on requiring flight demonstration of a high level of reliability with a high confidence level. Instead, a program philosophy such as that used in Project Mercury, with reliance placed on more subjective judgments, is to be expected. In this circumstance, judgment must be based primarily on a knowledge of the soundness of the fundamental design and on the quality of the workmanship. A conservative design philosophy, as used herein, implies that there is a high expectation of an early achievement of an acceptable level of reliability.

C. System Description

The injection vehicle system and its industrial support complex, as examined in this study, include the vehicle, its means of production, the associated transportation complex, and the facilities needed to assemble, check, and launch the vehicle (see Fig. 13). The spacecraft is considered only as it affects the injection vehicle system. These aspects of the system are briefly described in the following sections.

1. Injection Vehicle

a. Description and operation. The injection vehicle, which consists of four steps of clustered solid-propellant motors, is shown in Fig. 14. (Conceptual designs of a liquid- and a solid-propellant *Nova* are included for comparison purposes.) The motors in a step are joined by intrastage structure; successive steps are joined by interstage structure which contains provision for positive separation. Each step has its own thrust vector control system. The fourth step carries a 130,000-lb spacecraft, the abort rocket system, the guidance, control, and telemetry systems for the injection vehicle, and a vernier propulsion system.

A typical sequence of operations for injecting a payload into a lunar trajectory would be:

1. Stage-one ignition.
2. Step-one burnout, separation, and shroud ejection.
3. Stage-two ignition.
4. Step-two burnout and separation.
5. Stage-three ignition.
6. Step-three burnout and separation.
7. Vernier velocity correction into parking orbit.
8. Coast.
9. Stage-four ignition.
10. Step-four burnout and payload separation.
11. Vernier velocity correction into lunar trajectory.

Since vernier velocity corrections are used, it is not necessary to terminate the thrust of the large solid-propellant motors. The verniers required are small rockets containing about 5,000-15,000 lb of either liquid or solid propellant. Their thrust termination is within the state-of-the-art.

The desire for flexibility in the injection vehicle with respect to requirements originating with other missions might dictate the use of a storable, restartable, liquid-propellant fourth step. Detailed consideration of this possibility is beyond the scope of this particular study.

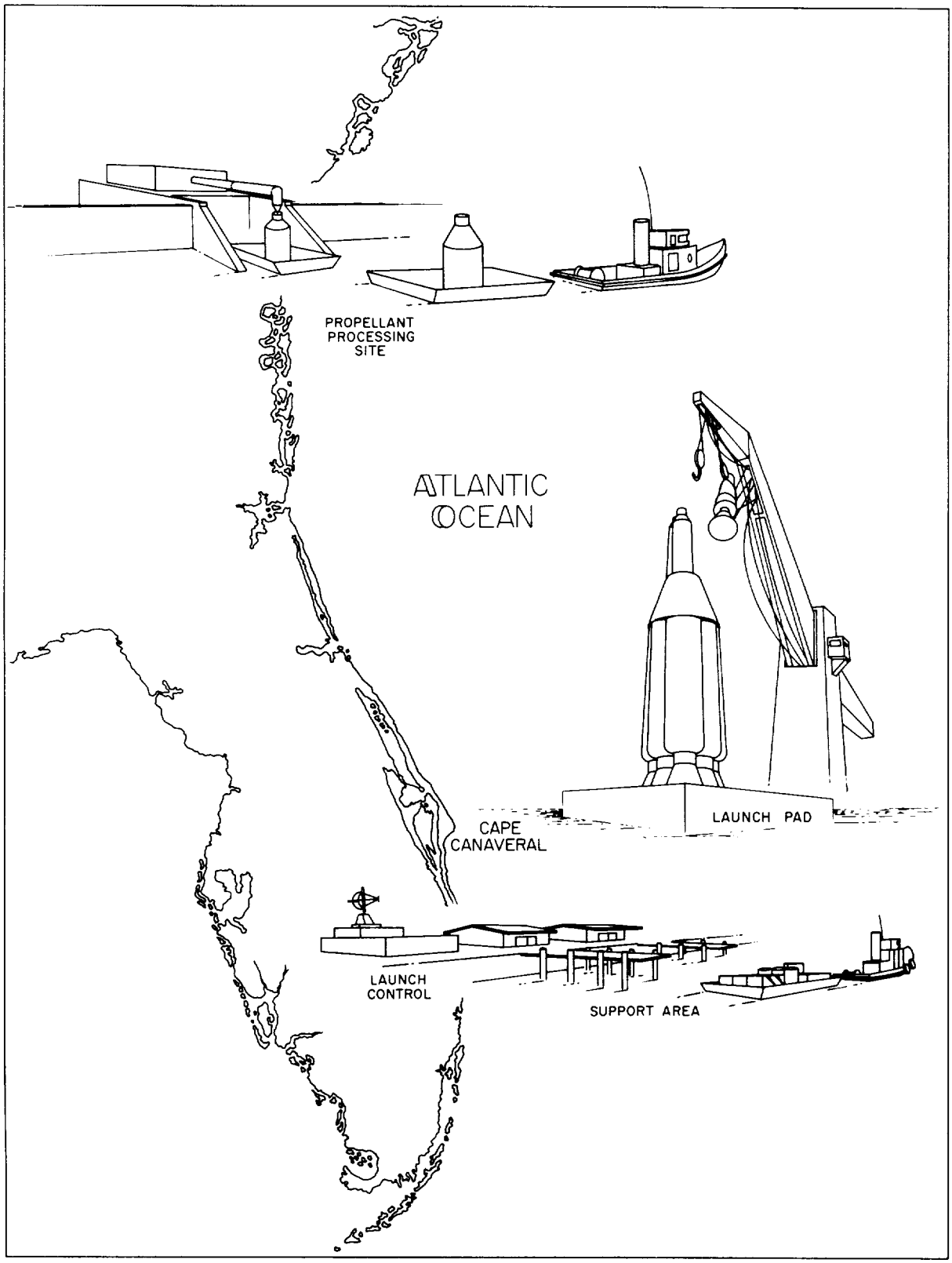

Figure 13. Solid *Nova* system

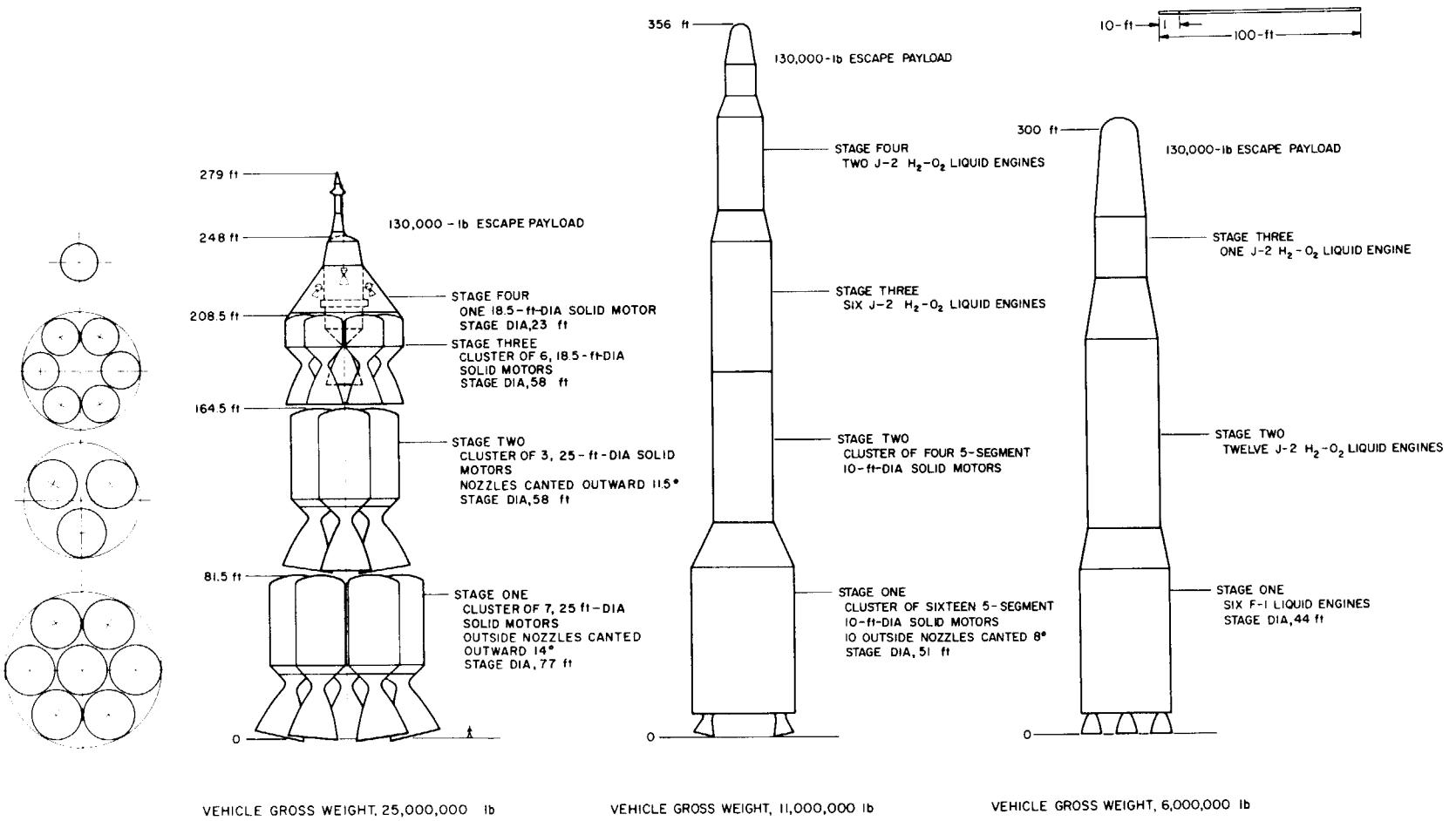

Figure 14. Conceptual *Nova* designs

The injection vehicle sized for the mission considered here has a gross weight at takeoff in the 25,000,000-lb class. The diameter of the first step is 77 ft, and the height of the injection vehicle to the separation plane between the fourth step and the payload is 220 ft.

b. Performance. Performance parameters are presented in Table 19. Three-dimensional point-mass trajectories were computed assuming eastward launch from AMR. The first two stages were flown gravity-turn after a short vertical ascent. The third stage was flown both at a constant angle in inertial space and with a constant inertial pitch angle. The latter mode produces a higher performance in going into a circular parking orbit at 100 nautical miles. It should be noted that the maximum acceleration, occurring near the end of third-stage burning, is 5.3 g, an acceleration tolerable by man. The mass will be distributed between the third and fourth stages such that the velocity at the end of third-stage burning is slightly short of achieving circular velocity. Part of the propellant weight assigned to the fourth stage is used as a vernier control to make up this difference.

The sizing was accomplished by assuming values for the specific impulse and for the step propellant mass fraction (i.e., propellant to gross step mass). The primary considerations used in the sizing and trajectory design were performance capability, airloads, achievement of a parking orbit at the end of third-stage burning, and meeting manned acceleration requirements.

c. Propulsion. The motors designed for this study represent a typical design concept to demonstrate the feasibility of making very large units. Detailed characteristics are shown below. Motor A is used for the first and second stages, and motor B is used for the third and fourth stages.

	Motor A	Motor B
Average vacuum thrust, lb	6,400,000	740,000
Propellant weight, lb	1,900,000	350,000
Liner weight, lb	15,000	7,000
Diameter, ft	25	18
Over-all length/diameter	3.26:1	2.33:1
Grain perforation	star	star
Volumetric loading, %	82	88
Burning time, sec	85	138
Nominal chamber pressure, psia	800	350
Burning rate, in./sec	0.64	0.46

A specific impulse of 245 lb-sec/lb at 1000 psi and sea level optimum expansion was assumed. The performance parameters assumed here are all within the present state-of-the-art. The reproducibility of performance should exceed that of present, relatively small motors because continuous mixing techniques are contemplated, and large amounts of propellant are being cast.

It is believed unlikely that combustion instability will be encountered; if it does occur, modern techniques should preclude any serious delay in schedule. In recent years, powdered aluminum has been found to be an excellent suppressant for combustion instability in existing solid-propellant boosters which use ammonium perchlorate composite propellants. As aluminum quantities are increased up to 15–17%, its performance as well as its effectiveness as a suppressant increases. None of the relatively large motors that utilize these high aluminum composite propellants (*Minuteman*, *Polaris*, and *Pershing*) have shown any sign of combustion instability; the propellants under consideration here contain these high aluminum concentrations.

Table 19. Vehicle performance parameters

Parameter	Stage			
	1	2	3	4
Average thrust, lb	44.6 x 10^6 (vacuum) 36.7 x 10^6 (sea level)	19.1 x 10^6 —	4.46 x 10^6 —	0.74 x 10^6 —
Specific impulse, sec	281 (vacuum) 231 (sea level)	281 —	294 —	294 —
Step propellant fraction	0.87	0.87	0.91	0.91
Number of motors	7 type A	3 type A	6 type B	1 type B
Total burning time, sec	85.3	85.3	138.3	138.3
Initial thrust-to-weight ratio, g	1.6	2.1	1.8	1.6
Maximum thrust-to-weight ratio, g	3.8	5.0	5.3	4.0
Velocity (ideal), fps	7040	8550	12890	10780
Velocity (burnout), fps	3800	12,800	25,460	36,140

Unitized motors were used in this study in order to investigate the ground facilities required to support this type of design. It is not necessarily recommended that they be chosen over segmented motors. However, since new propellant processing facilities will be constructed for this vehicle, they can be designed to accommodate either approach. The final decision between unitized or segmented motors should result from a study of the effects of the choice on the flight vehicle. Experience has shown that it would be undesirable to compromise the flight vehicle because of ground support equipment requirements unless a question of feasibility is involved.

Propellant will be processed in a new continuous-mix plant at a site strategically chosen for the raw material supply, its proximity to the launch site, and practicability of shipping finished rocket motors. Propellant facilities similar to those under consideration are already in production and have demonstrated the quality of product and high production rates required for the program. Aerojet-General Corp., for example, recently cast a 100-in.-diameter charge, weighing about 100,000 lb, at the indicated rate, then cured and fired it successfully. It is of interest to note that the static test record, based on high-frequency-response instrumentation, gave no indication of combustion instability.

d. Thrust vector control. A thrust vector control system is needed in each stage to compensate for the effects of center-of-gravity displacement, thrust misalignment, and unequal thrust (particularly at ignition and burnout). It must also counteract aerodynamic forces during first-stage burning. Canting of the rocket motor nozzles so that the thrust vector is directed near the burnout center of gravity of the stage helps to reduce some of these disturbances but cannot be used when the angle of cant is large. Thrust vector control systems that could be used include jet vanes, secondary injection, auxiliary rocket motors, and jet tabs. Each has distinct advantages and disadvantages, and a choice can result only from a detailed system study considering guidance, structures, and spacecraft. A jet vane system offers a feasible solution to this problem. However, a gas pressurized secondary injection system appears to be inherently more reliable and was chosen for more detailed study in order to provide a basis for weight and cost estimates.

e. Structures. The largest structural weight item in the vehicle is the rocket motor case. Fortunately, considerable experience in the design of cylindrical pressure vessels is available, and the large size considered here should introduce no problems which differ basically from those already encountered elsewhere. Consistent with the philosophy already presented, a heat-treatable martensitic steel with a yield strength of 165,000 psi was chosen in this study. Toughness and ductility should be high at this strength level. However, this conclusion is tentative until more information is available for the ¾-in.-thick material considered for the type A motor case.

The interstage structure, as presently conceived, is either a space frame or a set of three braced columns. Loads would be transmitted to the motor cases as concentrated loads acting on truss pads attached to the motor domes or as concentrated line shears. This technique was used on the *Sergeant* motor case and is extensively used in large water tanks and pressure vessels in the power generating and chemical industries.

Two types of rocket nozzles were considered: (1) a graphite-steel, heat-sink type, and (2) ablating plastic nozzles. The use of ablating nozzles appears very attractive for this application because the linear ablation rate will be the same or slightly less than that of smaller engines, and, therefore, the percentage change in throat area will be acceptably small. Either type appears feasible; scaling laws predict less severe conditions than for existing nozzles despite the relatively long burning time, provided that weights are scaled with impulse.

f. Structural dynamics. Major structural dynamics effects such as over-all dynamic loads and dynamic stability were examined qualitatively to ascertain whether extrapolation in size or weight would adversely affect the feasibility of the solid *Nova*. By using aeroelastic model theory, and by assuming that dynamic magnification factors associated with transit through discrete gusts are independent of size, it can be shown that a large vehicle should encounter less severe loading in relation to its strength than a small vehicle which is dynamically similar.

Dynamic instability of the type characterized by adverse coupling between the autopilot and the elastic airframe can become a serious problem if a conventional flight-control sensor installation is employed. However, "transducer arrays"—e.g., a number of attitude gyros discretely positioned over the length of the vehicle with outputs electrically summed—can be employed to suppress the coupling of the autopilot with body bending modes. Thus, low bending mode frequencies, *per se*, need not lead to dynamic stability problems.

The solid-propellant vehicle system carrying liquid payload rockets or a liquid secondary injection system poses relatively trivial liquid sloshing problems because of the low percentage of liquid mass.

The solid *Nova* development philosophy advanced herein places a major reliance upon the employment of scale models. In the structural dynamics area, it is considered that effective support can be given to the design effort and to the subsequent evaluation of the finalized design by recourse to model tests. The theory of aeroelastic model design is well developed, as are techniques for building such models. In the subject application, freedom from major liquid sloshing problems permits practical attainment of dynamic similitude in model design and test. Construction of, for example, a tenth-scale dynamic model for modal vibration surveys would permit simulation of local structural details such as bolted joints and use of dummy propellant having the proper density and viscoelastic properties. Smaller-scale aeroelastic models for wind tunnel test may be compared, through modal vibration tests, with the one-tenth-scale "reference" model.

g. Aerodynamics. The need to aerodynamically shroud the vehicle has been examined from the aspects of heating, drag, and unsteady flow effects.

Maximum laminar and turbulent heating rates were examined for the velocity-altitude information obtained from a powered flight computer trajectory. A conservative analysis indicates that heating will be no problem on exposed structural members or motor cases.

An estimate of the aerodynamic drag for a body of this type is not readily calculated. The examination of some experimental data and calculations based on two simple drag models indicates that the peak drag-to-thrust ratio is approximately 0.15, an acceptable value.

Unsteady flow effects between motor cases may result in high local vibration loads. Local fairing should alleviate this condition. In none of the cases studied could a positive requirement for shrouding be determined. Consequently, only a shroud from the payload to the top of the fourth stage has been indicated. The weight of this shroud was charged to the first stage, since it would be discarded at the end of first-stage burning.

The maximum dynamic pressure expected is approximately 1400 psf, and at first-stage separation the dynamic pressure is approximately 40 psf. These pressures are acceptable for the vehicle under consideration. Trajectory shaping can be used to reduce the maximum pressure to a value less than 1000 psf, if desirable.

h. Assembly and alignment. With proper attention paid to details, assembly and alignment should provide no difficulty. Provision will have to be made for a temporary framework to support the motors of a stage during construction until the stage is structurally tied together. Erection loads on the individual motor cases will have to be considered in their design. Vehicle loads caused by wind and wave action during storm conditions while on the launching pad are expected to be small for the blunt, dense, solid-propellant vehicle, and temporary guys or bracing are adequate protection.

Techniques are available to provide center-of-gravity control without measuring absolute weight. If further study indicates that accurate absolute weight measurement is required, an increase in the existing capability for accurately calibrating load cells is required. Load cells of the necessary size are available.

i. Guidance requirements. The guidance of the vehicle has not been examined in detail, but it is felt that no unusual problems will be present. The general philosophy of this program can be applied to the guidance area, and adequate space with a controlled environment and ample weight for the required equipment have been provided. The weight should allow for underrating components and providing redundancy wherever needed.

Since it is considered that midcourse correction capacity will be required in the spacecraft, the precision and performance required of the injection vehicle are not extreme. Indeed, guidance capability equivalent to that required for military weapons is adequate.

It should be noted that the general concept of launch, injection, midcourse, and terminal guidance of the vehicle as well as its spacecraft, controlled in part from the Earth,[2] is considered to be appropriate for this mission. Participation in the control by men on board should be limited to emergency measures only.

j. Design conclusions. Weight estimates for all of the steps in the injection vehicle have been based on the technical studies described above. As a result of this study it is concluded that a four-stage solid-propellant injection vehicle in the 25,000,000-lb class is feasible and is capable of performing the desired mission.

2. Development Plan

Consistent with the philosophy advocated in this study, the following development plan is proposed. Carefully planned, limited-purpose, small-scale tests will be used as a design tool to illuminate problem areas and to provide design tools or measures of reliability and add a

[2]Midcourse guidance is preferably although not necessarily controlled in part from the Earth.

measure of confidence and refined technical information to the development program.

Small-scale testing will satisfy most, if not all, of the requirements for design information within the first two years of the program. In fact, certain tests should be initiated along with design study in specific problem areas which can be defined early in the program. The small-scale test program will include:

1. Motor cases tested under pressure, concentrated loads and combined loads to provide design information and to confirm design techniques.
2. Prototype scale vehicles tested statically and dynamically to provide information on interaction effects, dynamic behavior and ultimate strength.
3. Wind tunnel models to provide rigid body and aeroelastic aerodynamic information.
4. Motors static fired to provide information on the charge design, nozzle, case lining and insulation, and thrust vector control.
5. Ignition tests, performed on a motor with a small web and the full-scale internal propellant geometry to provide design information.
6. Thrust vector control system tests to provide information on interaction effects and design information.
7. Miscellaneous tests on other structure, separation devices, and ancillary equipment.

The full-scale test program is relatively small and consists of:

1. Motor cases tested under pressure, concentrated loads, and combined loads.
2. Type A and B motors static-fired with active thrust vector control.
3. Miscellaneous subsystem elements such as the separation joint and some of the elements of the interstage structure.

It is important to note that the program approach and philosophy obviate the need for much of the conventional development and test program including all full-scale cluster static firings, full-scale structural dynamic testing, and boiler plate or partial vehicle flights normally associated with a vehicle program. These deletions result in a large saving in over-all program costs and development time.

3. Producibility

The producibility of solid-propellant engine processing facilities and raw materials, engine hardware fabrication including facilities and materials, and interstage and intrastage structure has been considered. It is assumed that rigid quality control would occur throughout all stages of manufacture and assembly of the vehicle.

Most of the manufacturing problems are associated with the large motor cases. Figure 15 shows some of the possible methods of manufacture. Material supply is assured, since the program under study would require less than 1% of the nation's current production of low alloy steel. However, new fabrication facilities will be required. It is recommended that they be located adjacent to navigable water because of the size of the manufactured items.

About one million square feet of space is required. A shear spinning machine and a gantry furnace with associated utilities are the only major additional items required. The rest of the metal hardware needed, although large, should provide no greater difficulty in terms of facilities or techniques than will the cases.

4. Facilities and GSE

Because of the size and nature of the program, some specialized facilities and operations are needed. These are examined briefly in the following material.

a. Propellant processing facility. Because of the quantity of propellant needed and the size of the loaded motors, water transport at the propellant processing facility is required.

The site should (1) be convenient to Cape Canaveral (the assumed launch site) by barge transport, preferably through an inland waterway to avoid long exposure to the open sea, and (2) provide in-plant barge transport during processing.

There are many islands and coastal areas from Texas to South Carolina that could meet the above requirements. A typical site, Skidaway Island, in Georgia (Fig. 16), has been chosen only to demonstrate the feasibility of such an approach. It is assumed that the island is undeveloped and that all waterways used during processing will be dredged completely. A description of the facilities is given in Table 20.

The propellant materials considered are typical components of propellants now used in large motor programs. New production capability is required for all of the ammonium perchlorate used in this program (8,000,000 lb per month). Power requirements for this production

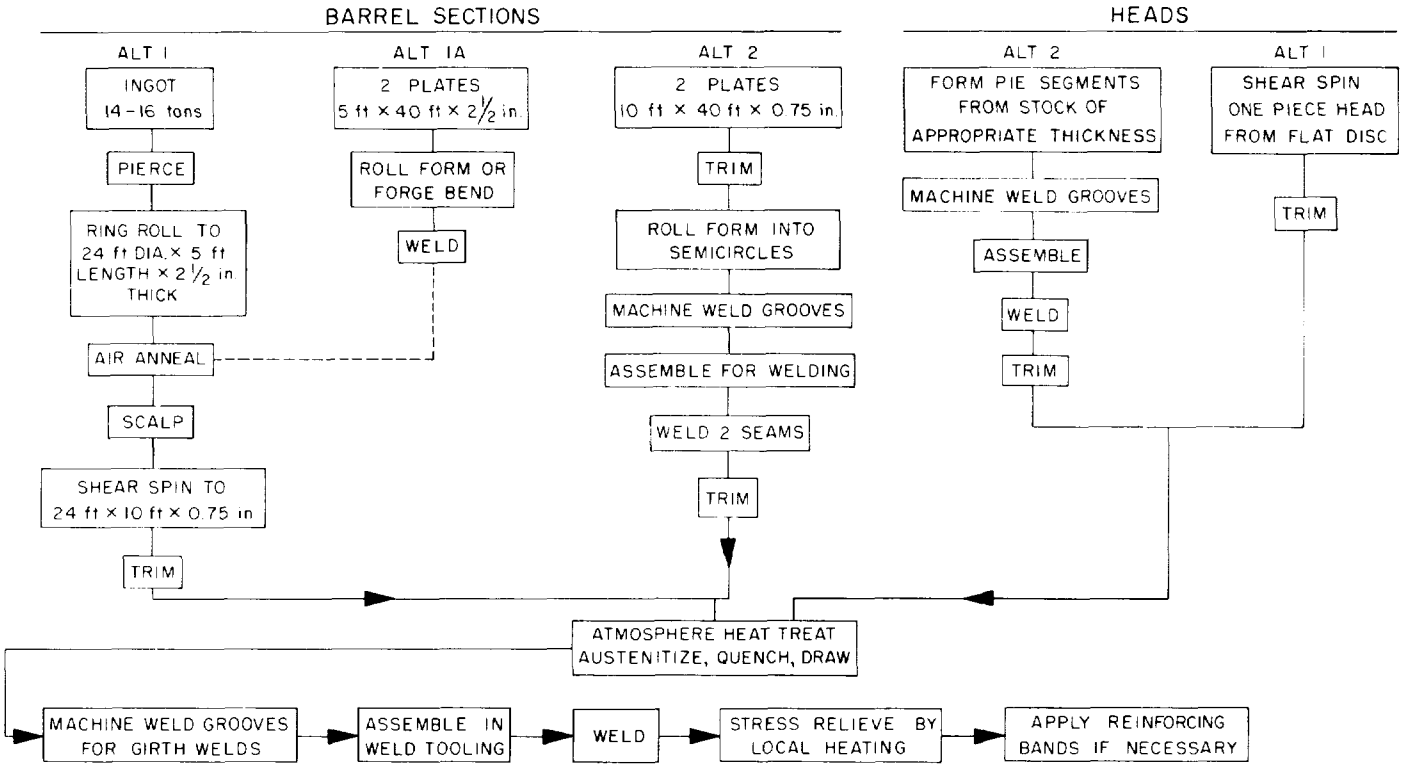

Figure 15. Possible manufacturing methods

are not excessive. The other materials required are readily available or could be made available on relatively short notice.

Propellants of the type under consideration, at the operating temperatures to be used, have never been known to detonate. Nevertheless, all facilities have been sized and sited on the basis of class 9 or 10 propellant because no experience at this size is available.

b. Launch site operations. The launch site, ground support equipment, and assembly and transportation techniques are governed by the following criteria: (1) the complex must be in operation within 2.5 years of the commencement of the program if it is to be used for motor static test firings; (2) it must be an economical and practical system for accomplishing this task; and (3) the possibility of loss, because of a launch failure, of costly and long-lead-time ground support equipment must be minimized.

The launch site is assumed to be located near existing range facilities and sources of manpower. The use of Cape Canaveral with its complete range system and support services as well as its years of extensive use is assumed here to be the chosen site. It is evident that the solid-propellant *Nova*, because of the acoustic and explosive safety distances involved, cannot be launched directly from the Cape. The choice of an offshore launching pad is indicated when the additional launch complex requirements are considered.

In determining support systems for the assembly and launching of a vehicle of this size, it is useful to observe the experience which exists in other fields of endeavor. Normal operations in the large civil engineering industry and in marine and naval architecture closely parallel the erection and handling techniques required for this vehicle. It is from this existing technology that the optimum solution should be derived.

Fixed underwater supporting structures for the depths required are encountered in bridge foundations and dam construction. Shipment of the weights required occurs daily in normal tug and barge operations. The erection and assembly of large pieces of equipment is within ship building and repair technology. Indeed, the largest man-made moving objects are ships of one form or another.

Thus, the optimum launch complex to meet the stated objectives is a fixed offshore launch pad, supported by vessels and barges and utilizing the Cape Canaveral range and support facilities.

c. Transportation and storage. Loaded solid-propellant rocket motors are stored in their respective loading

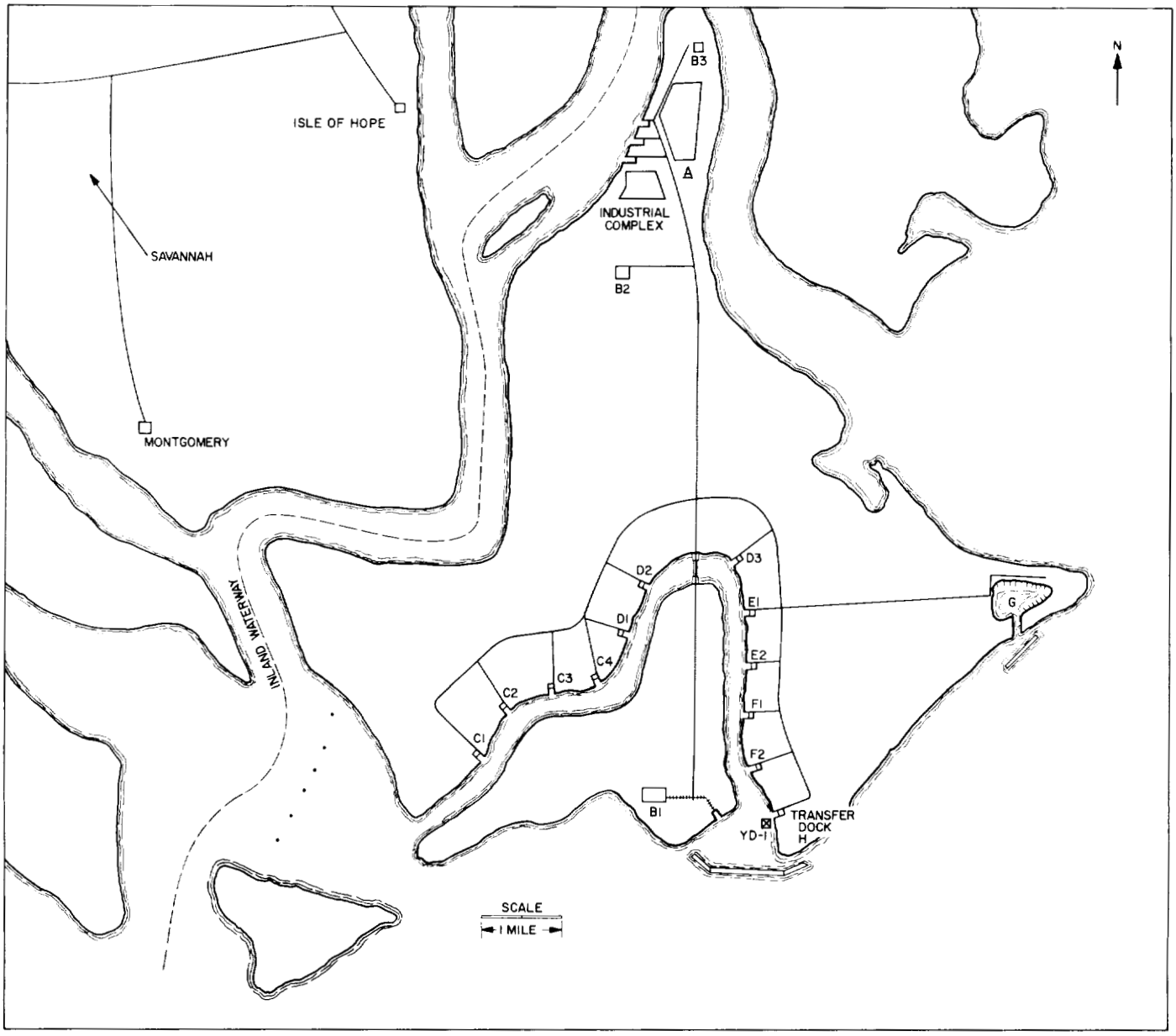

Figure 16. Propellant processing facilities

barges at the propellant processing site. Motors are transferred from the loading barges to the transport barges by the floating crane at the processing site. They are then tugged to the launch site at a rate determined by the launching schedule.

Guidance, telemetry, and associated equipment is transported to the launch support area at the Cape by conventional means. After checkout and maintenance, the equipment is barged to the launch pad for installation.

d. Site construction. Since vehicle assembly and checkout procedures require four months (Fig. 17), two launch pads are required to allow launchings on two-month centers. These pads (Fig. 18) would be placed approximately 6 to 12 miles off the coast of Cape Canaveral with a similar distance separating them for acoustic and quantity-distance safety requirements. The launch pad would then be in 60 to 100 ft of water. This depth is typical of that found in bridge foundations, and this construction is the type desired. For a solid-propellant vehicle, the dimensions of the pad need only be large enough to provide structural support for the vehicle. The large weight of the vehicle as erected provides considerable stability against overturning caused by wind and

Table 20. Physical descripiton of facilities

Designation	Number of Units	Components of Each Unit
Facility A, case processing	1	Standard industrial buildings including mixers for preparing liners and insulation
		Equipment for weighing, cleaning, lining, and insulating the chamber
		Three receiving docks
Facility B1, storage of Class 2 materials	1	Storage buildings with revetments, drying and grinding units for ammonium perchlorate, one dock
Facility B2, storage of Class 1 materials	1	Storage tanks and buildings
Facility B3, engineering office building and raw material quality control laboratory	1	Office building, chemical laboratories
Facility C, mixing and casting of propellant	1	Fuel premix station, storage tanks
	4	Two 1.5 x 10^6 lb/month mixing plants, similar to the plant now in operation at Aerojet-General Corp; propellant is pumped over a boom to the motor
		One casting dock with corrugated rain shed
Facility D, curing	3	One curing dock with rain shed, power supply for heating mantles
Facility E, final motor processing	2	One assembly dock with rain shed
		Tooling for loosening and removing the mandrel, trimming the propellant, and mounting the nozzle
Facility F, inspection	2	One dock with more elaborate overstructure for inspection equipment
		Inspection units
Facility G, storage	1	A dredged lagoon with sufficient weather-protected docks for storage of motors for 2 complete vehicles
Facility H		Two transfer docks, floating crane

storm conditions. Hurricane protection is limited at most to temporary guying and bracing. All auxiliary operations emanate from the crane or support vessels. A hold-down structure is neither desirable nor practical for use during the launching of a solid-propellant vehicle.

A breakwater of one mile in over-all length is required to shelter each pad area sufficiently to allow shipborne operations in all but gale conditions. This breakwater is of rock construction and 10 ft above high tide.

General support, electronic checkout, engineering personnel, and auxiliary equipment receiving and storage buildings are located on or near the Cape. Several docks and a crane (YD-4) are required to transfer this equipment to barges for transport to the launch pads.

The distance off-shore between the pad and the Cape has been specified as that allowing for inhabited areas, according to the mass of propellant involved. The Launch Control Center is, therefore, not a blockhouse in the normal sense, but is structurally an ordinary building, located on the Cape in the general support area. This Center contains the launch control instrumentation and equipment and is connected to the pads by underwater cable for phone and electrical connections. Most of the prelaunch instrumentation and monitoring measurements are made, however, by direct radio link between the vehicle and the Launch Control Center.

An umbilical mast provides electrical connections to the spacecraft up until launch, as well as emergency de-arming or astronaut exit ladders. This mast pivots at its base and drops to the water at launch.

e. Vehicle assembly equipment. The primary assembly problem is the erection and handling of the first- and second-stage motors. A crane of 1,000-ton capacity and 200-ft hook height is required, with a secondary hook of 200 tons and 300 ft for the upper stages and payload.

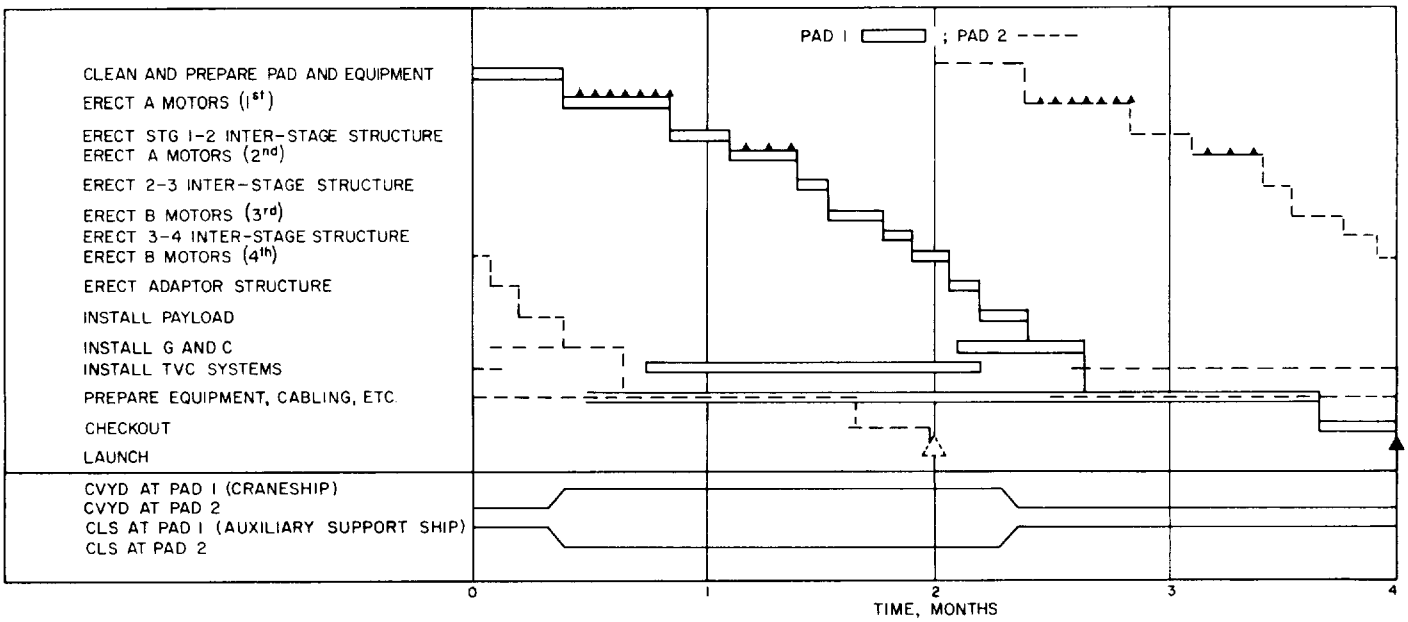

Figure 17. Launch operations schedule

The construction and operation of this crane and its support structure are considerably simplified when based on water-borne operations. Currently, the largest movable crane in the United States is the Navy YD-171, mounted on a self-propelled floating barge. This is one of four constructed in 1941 and has a 450-ton capacity and 160-ft hook height. Several fixed cranes of this capacity, but with lower hooks, also exist.

Construction of the crane required is considered quite feasible, and two options for its flotation are available.

A scaled-up version of the flat-bottomed barge and crane may be specially constructed for this purpose and built to contain personnel and service areas. A second method is the mounting of this crane on a demothballed World War II aircraft carrier (CVYD) of the Midway Class (Fig. 19). This method has several desirable secondary features. It supplies an essentially self-contained mobile field unit for personnel, power, and work area at a nominal demothballing cost. Prior to the YD-171, the largest naval crane utilized this same technique; it was constructed amidships on the Kearsarge, an obsolete pre-World War I battleship.

The use of a floating crane for the handling and careful positioning of very large loads is a well-established procedure. It provides the most economical and shortest-lead-time solution to the erection problem and allows for convenient removal from the pad area prior to launch.

Personnel service platforms around the vehicle are made in prefabricated sections and are attached directly to the vehicle structure. The loads introduced are small compared to the load-carrying ability of a solid-propellant vehicle skin. Personnel access is via an elevator tower on the craneship and gangwalks across to the vehicle. The platforms are removed prior to flight.

An auxiliary support ship (CLS) with an elevator tower for crew access is required. This ship provides checkout services and a nominal-capacity hoist for the removal of personnel scaffolding and small gear and test equipment.

f. Static testing. The full-scale static test firings of the individual full-sized motors can be conveniently accomplished at one of the launch pads. A motor, supported by suitable structure, can be mounted in an inverted position on the pad. Normal launch control instrumentation is used for this series of tests. The alternate pad is used for the first flight vehicle. Upon completion of the static test program, the external support structure is removed, and this pad is converted to a flight pad for the second flight firing.

Alternatively, a separate static test stand can be constructed and utilized for this purpose. The additional area and cost for this test stand are included in the cost analysis.

g. Field operations. A flow diagram of the field operations is shown in Fig. 18. Individual motors of types A

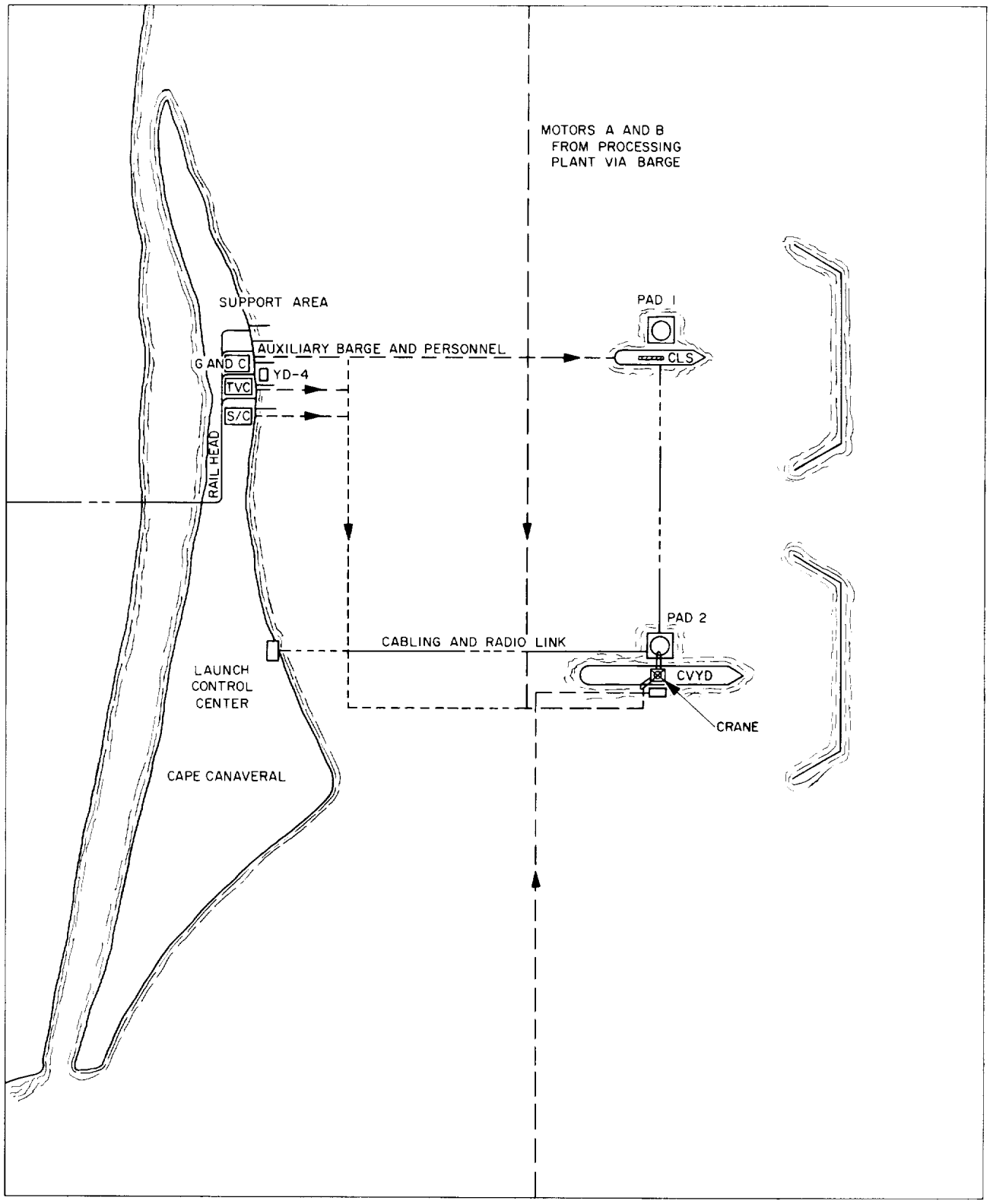

Figure 18. Launch and support complex

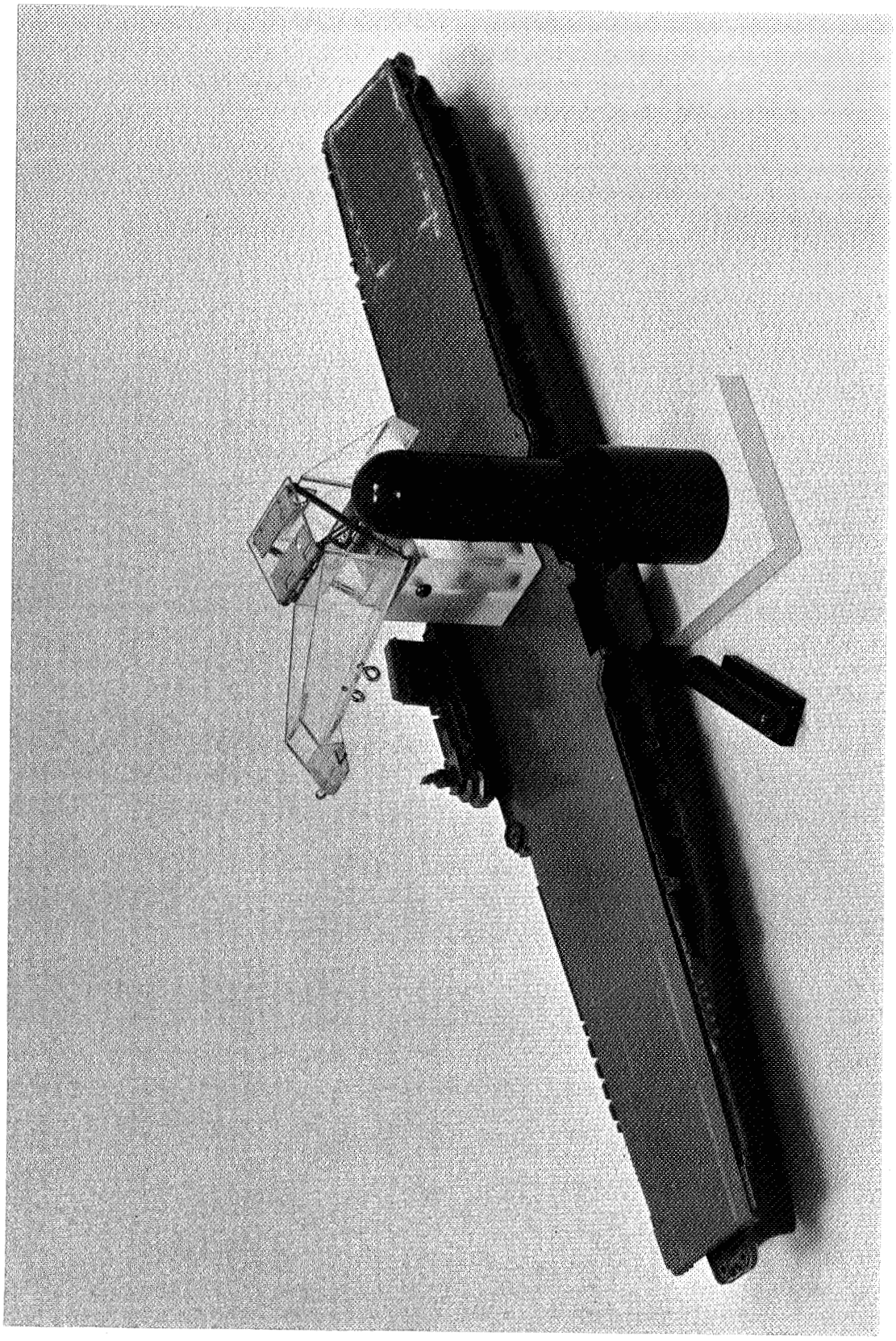

Figure 19. Pad assembly

and B arrive at the launch complex by barge from the propellant processing plant storage area. They are erected and assembled on the launch pad by the craneship (CVYD) or crane-barge. Interstage structure, in large prefabricated sections, is barged out from the Cape support area and erected. The payload and thrust vector control devices are also barged out and installed. The crane-ship is then transferred to the alternate pad and is replaced by the auxiliary support ship (CLS). During the remaining period the guidance and control, telemetry, and spacecraft equipment together with auxiliary gear is installed and the final checkout operations are accomplished prior to launch.

The firing of this vehicle from an offshore location at the Cape presents no extraordinary problems with respect to range procedure. Normal down-range tracking, range safety, and launch monitoring are identical to regular Cape launchings. Firing windows are reasonable and the actual final countdown time for the multistage solid-propellant vehicle is relatively short compared to that for existing vehicles.

5. Spacecraft Considerations

Spacecraft studies have been limited to crude feasibility determinations based on (1) 130,000-lb injected weight, (2) configuration constraints, and (3) spacecraft design problems peculiar to the employment of the solid-propellant *Nova* for a manned lunar landing and return mission.

The results of the studies can be summarized as follows: (1) there are no spacecraft design problems peculiar to the solid-propellant *Nova;* (2) the injection capabilities of the solid-propellant *Nova* appear to be adequate for a manned lunar mission; (3) there are no major spacecraft configuration constraints or limitations due to the injection vehicle.

For the purposes of this study, *Apollo* three-man mission and command modules have been arbitrarily assumed. Mission abort capabilities are assumed to be consistent with the guidelines established by the Space Task Group.

It is recommended that the spacecraft be capable of accomplishing the entire mission automatically. Men would perform monitoring functions associated with the control loop and would help implement scientific measurements and observations. Manned override control capabilities would be provided for emergencies to maximize mission reliability for the manned phases. A possible feature for a manned mission would be prior provision of an alternate return vehicle on the surface of the Moon as a contingency for possible failures during landing. In this event, man would be equipped to transport himself over the lunar surface from one vehicle to another.

Of all the spacecraft environments associated with a solid-propellant *Nova* (vibration, linear acceleration, acoustical, etc.), the one which appeared to be most severe when compared to a liquid-propellant vehicle of the same capability is the acoustical environment. Differences in the other environments are minimal.

An analysis of the acoustical environment has been made based on extrapolations from data for smaller engines. However, the results are considered to be conservative. The calculated sound pressure levels are:

$$L_p\,(solid) = 167 \text{ db}$$
$$L_p\,(liquid) = 161 \text{ db}$$

These levels correspond to a distance of 200 ft from a solid-propellant vehicle with 40,000,000 lb of thrust or a liquid-propellant vehicle with 9,000,000 lb of thrust.

Actually, the lower exhaust velocity of a solid-propellant motor causes it to have a lower acoustic efficiency, so that the pressure level from a liquid-propellant vehicle might well be higher than that from a solid-propellant vehicle. In any case, it is important to note the high pressure level from either.

Factors such as sound absorption in the air (which increases at these high intensities because of nonlinear damping) and directivity of the sound should decrease the levels by 20 db. Reflection from the pad could be minimized by flowing water under the booster at liftoff. These factors suggest taking the pressure level as 150 db.

The ears of the astronauts are most sensitive to the acoustic field. The effectiveness of ear protectors is limited by bone conduction to about a 40-db reduction. Thus the pressure level at the ear will be reduced to 110 db, which is below the threshold of discomfort at 120 db and well below the threshold of pain at 140 db. The maximum total recommended level for speech comprehension is 110 db, so that talking with the astronauts during liftoff may be difficult, although additional attenuation by the cabin walls may make it reasonable.

It should be emphasized that the attenuation of the sound with distance is critically important.

D. Program Schedules

Figure 20 presents schedules for the over-all system including the vehicle, propellant processing, and launch

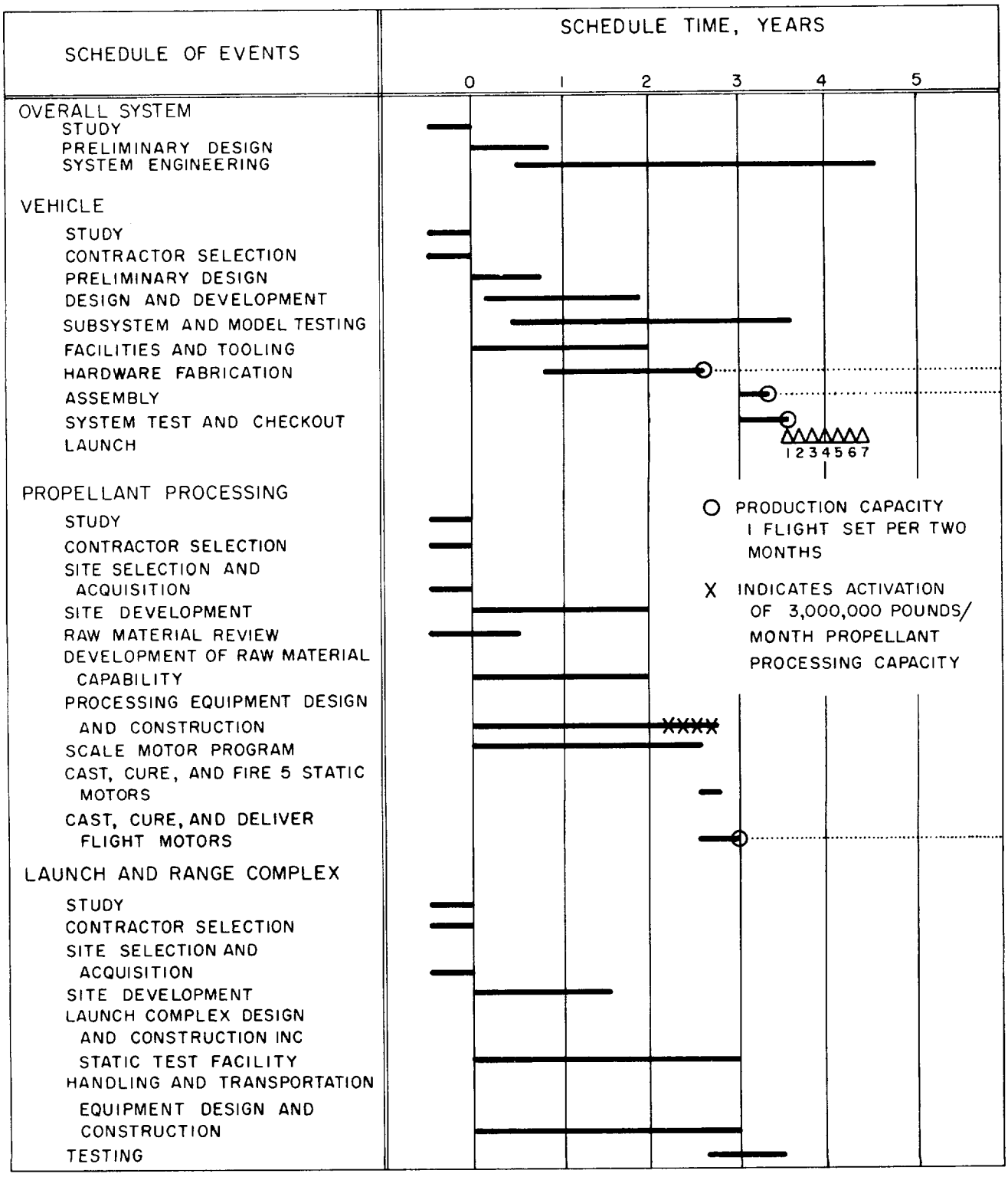

Figure 20. Program schedule

and range complex. The schedules shown are believed to be those which would be pacing items in the vehicle program. A spacecraft schedule is not shown because insufficient study was made to determine whether or not it is a pacing item.

Seven launch dates appear in the vehicle schedule. The first two are intended as propulsion rounds wherein the primary emphasis will be to evaluate the performance of the vehicles as a system. The next four are spacecraft flights. The primary objective of these four flights will be to gain knowledge and confidence in the various events which would ultimately occur during a manned lunar landing and return. It is believed that the successful accomplishment of the first six flights will result in sufficient knowledge and confidence to enable the seventh flight to carry men to the Moon and return.

There are a number of factors in a program such as is proposed which generally occur at the beginning and which are impossible to schedule at this time. As a result of this, the time 0 (zero) shown in the schedule is not a calendar date, but is a time which is dependent on completion of the following items:

1. The program shall be approved and funded to permit immediate pursuit of preliminary design with follow-on approvals and funding to occur so that schedule delays are obviated.
2. A minimum of six months of study shall have occurred prior to time 0 (zero), during which time the application of the solid propellant *Nova* to the lunar mission shall have been verified in depth and the technical requirements and preliminary design specifications generated.
3. Major contractor qualifications shall have been determined and major contractors selected. These contractors' personnel shall be available to start preliminary design.
4. Site requirements shall have been established, sites selected, and immediate availability established for launch complexes, propellant processing, loading, storage, and any other major sites required.

An important feature to be stressed is that a six-month period is devoted to study of the vehicle system in depth in order to specify the system and industrial complex requirements. During this period major and pacing contractors will be selected.

It is desired that contractors participate in the study if possible; however, it is required that key personnel from each of the major contractors participate in the nine-month preliminary design. As this work proceeds it is expected that the contractors will make commitments to carry out their portion of the program.

In addition to the schedules which have been presented, it is felt that some comments which compare inherent schedule characteristics of liquid- and solid-propulsion system development programs are warranted. It is believed that the comparisons which are made justify the belief that an inherently shorter development time is required for a solid-propellant propulsion system than for a liquid-propellant propulsion system with the same reliability.

1. A solid-propellant motor is comparatively simple. However, it has limited capability; that is, it is difficult, if not impossible, at present, to use a solid-propellant motor when there are requirements such as restart and throttling. Liquid-propellant systems are at present more complicated. However, restart and throttling are rapidly achieved. As used in the proposed system, a complicated system is not required, and the inherently simpler solid-propellant motors suffice for the job.
2. The interaction of the propellant in a solid-propellant motor with its flight environment is practically negligible. That is, there are no internal ballistics problems which can occur in flight that are not revealed by static testing. This usually results in a higher reliability with the solid-propellant vehicle at a given early date in the flight development program.
3. In a solid-propellant system, the majority of subsystems can and must be developed concurrently, and interactions between subsystems are usually minor. It is fairly typical in a liquid-propulsion system that the development of subsystems is carried out in a sequential manner, and the interactions between subsystems frequently cause multiple iterations during the development.
4. Size scaling has been repeatedly demonstrated in ratios in excess of that required for the proposed solid-propellant engine program. Size scaling is more uncertain with liquid systems.
5. Insofar as the propulsion units are concerned, there are few operations which need to be checked at the launch complex. Thus, the launch complex and the ground support equipment are inherently much simpler than are the launch complex and GSE for a liquid system.

6. In a liquid-propellant system, it is necessary to perform many operations in flight such as the metering in total quantity and proper proportions of propellants which are to be burned. In a solid-propellant motor, these operations are performed in the factory.

E. Program Costs

It is expected that in 5 years this program will allow the successful placement of a man on the Moon and return, at which time approximately $1.9 billion will have been spent.

Costs have been estimated (Table 21) based on a flight program of 20 rounds extending over a time period of 7 years. For this program the total cost is approximately $2.6 billion. Included are vehicle and spacecraft development and production costs, all special production facilities such as the propellant processing plant, launch facilities and GSE, and launch operations. The astronaut training program is not included. Launch from an offshore pad at AMR is assumed.

1. Vehicle Costs

The major portion of the study effort was directed toward the injection vehicle. Cost studies were also aimed principally at the vehicle. Both development and production costs have been estimated.

To understand such features as development time, reliability, and cost of this boost vehicle program, it is necessary to have the underlying concept of the injection system clearly understood. This concept states that the inherently simplest and most reliable rocket motor shall be merely scaled up to provide the desired injection capability. Complication or advancements in the state-of-the-art shall not be introduced to save weight or increase performance. Application of this concept has many important side benefits. Among these benefits is a comparatively small development program and very little full-scale testing. This fact, along with the inherently

Table 21. Cost summation

	Development, $ millions	Production (based on 20 vehicles), $ millions
Vehicle		
Airframe and motor cases	87	221
Rocket motor propellant and processing	60	440
Auxiliary equipment	73	31
Guidance, communication, and power	90	66
Launch operations	—	27
Launch and support personnel	10	105
	320	890
Facilities		
Propellant processing facility	115	
Ammonium perchlorate plant	30	
Motor case and nozzle facility	115	
Launch pad and GSE	110	
Launch base	33	
Static test facility	15	
Transportation and storage	23	—
	441	
Injection system total	761	890
Spacecraft		
Command and mission module	500	140
Landing and return bus	250	70
Propellant and miscellaneous		10
	750	220
Program total — development	1511	
Program total — 20 flight program		1110
Total — development and 20 flights	2621	

simpler production characteristics of a solid rocket, results in a low-cost vehicle program.

a. Airframe and motor cases. The development program for the primary structure consists largely of design engineering of the full-scale structure and a small-scale program for structural and dynamic testing. Only motor case pressure testing and concentrated load testing will be done at full scale. Production costs were computed at $4/lb for motor cases, $10/lb for nozzles, and $5/lb for interstage and intrastage structure.

b. Rocket motor.

1. Motor development. The major elements of the rocket motor development program are a subscale development program for the two rocket motors and a full-scale program of five static firings for each motor discussed earlier under Development Plan (Section C-2). No full-scale captive firings of a cluster are planned.

2. Propellant processing. Eight continuous process lines similar to the one currently in operation at Aerojet-General Corporation are required to provide sufficient capacity for the proposed firing rate. Based on costs from the operating line and allowances for the larger scale motors, an operating cost of $1/lb of propellant was estimated. This figure includes raw materials, case preparation and lining, mixing, casting, curing, final assembly, and inspection. Costs for all plants and facilities, motor case and nozzle, and transportation are included in other items in Table 21.

c. Auxiliary equipment. Auxiliary equipment includes thrust vector control, stage separation, and thrust termination or vernier velocity control systems.

Thrust vector control could be achieved through secondary fluid injection by a simple pressure-feed system. Component development costs for this type of system were computed at $10,000/lb. High-pressure tankage development costs were estimated at $8 million.

Stage separation development, design, and testing would cost approximately $5 million. Vernier velocity control system development would cost $7 million.

Production of high-pressure auxiliary tankage, feed system components, separation devices, etc., costs approximately $200/lb.

d. Guidance, communication, and power. The injection guidance problem was assumed to be equivalent to that for a liquid-propellant rocket vehicle from a cost point of view. The injection vehicle guidance, communication, and power system weight is estimated to be 3,000 lb. Development costs of $30,000/lb and production costs of $1100/lb were assumed. This allows for environmental control and redundancy in accordance with the basic philosophy advocated.

e. Launch operations. To sustain the six-per-year firing rate it is estimated that 1,000 people are required continuously at the launch base. This includes launch pad assembly and firing crew, hangar checkout and assembly crews, and all technical support personnel at the base. The cost of maintaining this force is estimated to be $30 million per year. In addition, transportation of motors from the storage site of the processing plant to the launch pad will cost $350,000 per vehicle (program total of $7 million), and additional launch operations will cost $1 million per vehicle (program total of $20 million). Costs for use of the range are not included.

2. Facilities

General facilities such as engineering space, shops, and laboratories have been included in the development and production cost estimates. Such facilities are provided when (1) sufficient capacity does not exist to meet program needs, (2) existing facilities are not large enough to accommodate the size, or (3) location at some peculiar site is required.

a. Propellant processing facilities. Starting from a totally unprepared site on a waterway, the following facilities are required: case processing, storage and preparation of ammonium perchlorate, storage of other propellant and liner materials, engineering offices, raw material quality control laboratory, mixing and casting of propellant, curing, final motor processing, inspection, storage for enough loaded motors to make two complete vehicles, transport dock and crane, and island development. The cost does not include the purchase of the real estate itself, approximately 60 square miles, since that depends on the final site selection. It does include the complete island (or waterway site) development in the form of land clearing, feeder access roads and utility installation, complete dredging of the internal canals and basins, and all excavation, roadwork, and land preparation.

It is assumed that a processing site can be located sufficiently close to a population center to provide the required work force of approximately 2,500. No costs have been included for housing or support of this work force.

b. Ammonium perchlorate plant. The present national capacity for production of ammonium perchlorate is insufficient to meet the program requirements. This being a captive industry, the cost of a wholly new facility capable of producing 8,000,000 lb per month must be borne by the program.

c. Motor case and nozzle facilities. Because of the large size it is assumed that a motor case and nozzle fabrication and assembly facility must be constructed in its entirety.

d. Launch pad and GSE. Two offshore launch pads with one blockhouse are required at AMR. A breakwater must be provided for each pad. Floating cranes and support ships are used for assembly and launch.

e. Static test facility. No full-scale static structural or dynamic testing is planned on the complete launch vehicle. Adequate facilities already exist for structural static and dynamic testing planned on the subscale models.

Full-scale rocket motor tests of the individual motors will be conducted on the launch pad. Modifications to the pad for this purpose will cost $5 million. A separate static test facility would cost approximately $15 million. Facilities currently exist to handle both motor subscale firing programs.

f. Transportation and storage. The first- and second-stage motor unit creates the major transportation problem. Barge transportation is clearly the only feasible method for empty or loaded motors. It was assumed that the processing plant would be located along the East or Gulf Coasts. Enough barges were included to provide storage between loadings and firings. The case fabrication facility must also be located at a major waterway.

No other elements of the vehicle present any transportation problem of significance.

3. Spacecraft

The spacecraft consists of two major systems: (1) the command and mission module containing the cabin and crew, environmental control, life support, communication, guidance, and reentry equipment and protection, and (2) the lunar landing and return bus which also provides in-course guidance and maneuvers.

Apollo studies conducted by Martin and Convair were used to obtain command and mission module costs. Development costs of $50,000/lb and production costs of $700/lb have been used for the module. The landing/return bus costs were computed at $25,000/lb for development and $350/lb for production, which are typical costs for the currently envisaged dry-weight bus. These cost estimates cover the spacecraft developer's engineering cost and spacecraft procurement for the *Nova* phase but do not cover the flight test program for spacecraft development, prior to the *Nova* phase.

4. Conclusions

The cost study of the solid propellant *Nova* injection system has revealed some rather unusual, albeit tentative, conclusions. The conclusions reached are a logical consequence of the basic program philosophies.

1. The development costs for this vehicle are considerably lower than for a liquid-propellant vehicle of comparable capability.

2. Production costs per vehicle are roughly comparable to a liquid-propellant vehicle, depending somewhat on the injection mission and the total number of flights.

3. Injection costs in dollars per pound of payload for a 300-mile Earth orbit for a 20-vehicle program are $169/lb total cost and $91/lb production cost. For these computations a reliability of 95% and 515,000 lb of injected weight were used.

4. The development costs are low because the usual full-scale development support programs are not required – no full-scale vehicle structural testing, no captive firings of clusters, no "battleship" propulsion program.

5. Production costs per pound for the structure are low because of the inherent simplicity of a solid-propellant rocket. Size was simply exchanged for complexity at nearly constant total production cost.

F. Growth Potential

The importance and urgency of the manned lunar landing mission obviously focuses attention and major vehicle effort on quickly accomplishing this major-milestone mission. However, in the international space competition, the manned lunar landing and return is merely the first lap in a sustained race. If the nation is to avoid a deficiency in the next "big booster" requirement, some thought must be given to NASA's next major milestone, with a quick look at the following milestones. At any one time, it would seem judicious to have our space plans span at least two major milestones in order that technical and fiscal planning be sound, that continuity be maintained, and that intermediate or minor milestones reinforce one another.

It is assumed that the next major milestone will be a permanent manned lunar laboratory or base and that, in time, this would lead to the next major milestone, man-on-the-planets.

A cargo version of the *Nova* class spacecraft under consideration can place approximately 35,000 lb gross weight on the lunar surface. Such devices as Moon-mobiles, prefabricated structures, and life support systems can be delivered directly from Earth, intact and ready to operate with no assembly or disassembly.

The 25,000,000-lb solid-propellant *Nova* would appear to have considerable growth potential beyond this. Substitution of liquid hydrogen–liquid oxygen stages for the third and fourth solid stages, such that gross weight is unchanged, results in a vehicle that could place approximately 110,000 lb of men and equipment on the Moon. The first two very large solid-propellant stages would be used as developed for the manned lunar landing and return. The third stage might consist of a cluster of 12 J-2 engines with tankage modified to contain about 1,500,000 lb of propellant (possibly a cluster of three second-stage tanks from the *Saturn C-3*). The fourth stage could be a single S-II stage from the *Saturn C-3*.

This solid–liquid propellant *Nova* vehicle should be capable of placing approximately 930,000 lb into Earth orbit. If one were to use this weight as an electric-powered spacecraft with 50,000 lb of radiation shielding for a 3-man crew using 15 lb/man/day for sustenance, it should be possible to perform a Mars manned landing and return in approximately 620 days.

III. SUMMARY AND CONCLUSIONS

It is appropriate to summarize briefly the preliminary system studies, results, and conclusions and to indicate those steps which should be taken in the near future which would assure that the system selections to be made in initiating the formal development programs for the manned lunar-landing mission can be made with a maximum degree of confidence.

The solid-propellant launching-vehicle study assumed the use of solid propellant in all stages. Many of the conclusions reached are also applicable to the solid-propellant stages of mixed (solid propellant and liquid propellant) systems. The predominant conclusions which have been reached are as follows:

1. A very large, *Nova*, solid-propellant launching vehicle is feasible.
2. The spacecraft-system requirements for either solid or liquid propellant launching vehicles are essentially identical.
3. The injection-guidance requirements for either solid or liquid propellant launching vehicles are essentially identical.
4. Specifying a conservative level of performance for the rocket-motor specific impulse and metal-parts design should, by trading size and weight for time and money, result in an economical program with an early achievement of a useful level of reliability.
5. In these circumstances, the spacecraft development schedule, rather than the launching-vehicle development, would probably become the pacing item in the over-all program.
6. The use of segmented grains is not required for very large solid-propellant motors. The use of maritime equipment and operating techniques avoids this complicating factor. Further study may indicate propellant processing or interior ballistics reasons for preferring a segmented design.
7. The cost and schedule information is based on a minimum development program; the initial conservative specification makes it likely that this result can be more nearly achieved with this system than with systems which require more substantial technological developments.

The study of the use of rendezvous on the lunar surface is characterized, as are the related concepts of rendezvous in a near-Earth orbit or in a lunar orbit, by greatly reducing the size of the launching-vehicle development problem and by requiring the development of an operating capability in a non-terrestrial environment. A combination of the various rendezvous possibilities leads to the absolute minimum launching-vehicle requirement. (See the Lundin Committee report for considerations of these other system concepts.) The more important conclusions reached in the study of the lunar-surface (only) rendezvous system are the following:

1. The launching-vehicle performance requirement is met conveniently by the currently proposed *Saturn C-3* configuration. A minimum mission could be supported at the *Saturn C-2* performance level.
2. The principal system modules, for example, the landing and return bus, would have multiple uses during the system buildup and this factor contributes to an early buildup of a useful level of reliability.
3. The development of all components of the system can be started simultaneously. The inherent flexibility of a rendezvous concept makes it possible to accommodate a wide range of variations of final performance of the developed systems.
4. The lunar-surface operations must be carried out in a thermal environment which is more varied than that of near-Earth orbital operations; however, the *Surveyor* studies indicate that no substantial special difficulties are introduced.
5. The solid surface and the gravitational field should make lunar-surface operations substantially easier than those in a near-Earth orbit. Furthermore, meaningful development testing of this phase of activity can be carried out on the Earth's surface.
6. The system treated in the study does not have the additional propulsion capacity to provide for an emergency abort capability during the manned lunar-launching or takeoff phase in case of propulsion failure. This characteristic is not unique to the lunar-surface rendezvous system. None of the many systems which have been discussed are based on a scale which permits propulsion failures during the lunar-landing or takeoff phase.
7. Equipment failures during the buildup phase can be accommodated without compromising the over-all mission.

8. The buildup phase includes a number of lesser goals which would be of significance in both the technological and the public relations areas.
9. The entire mission can be performed in an automatic mode of operation. This relieves the crew and, in particular, the spacecraft commander of unnecessary stress and permits the primary attention to be placed on mission status and the need for the use of alternate procedures.

In addition to the industrial study efforts on the *Apollo* spacecraft now initiated in response to the current RFP, there are a number of other activities which would or could help to clarify the over-all system-evaluation problem in the near future.

The advanced-development program for large solid-propellant rockets has resulted in a very successful static test of a 100-in. diameter segmented design by Aerojet-General Corporation. Additional test firings in this program are scheduled which will provide more information on the development problems associated with large solid-propellant vehicle-systems.

The lunar-surface rendezvous concept could be substantially clarified by a sponsored industrial study effort on the transit, landing, and return bus. A tentative plan exists for such studies as part of the *Prospector* mission and the studies could be initiated in the near future. If suitable attention were given to interface problems, such a study could be equally applicable to the *Prospector* mission and to the manned lunar-landing mission. In fact, if lunar-surface rendezvous were to be adopted as a prime system for the manned lunar-landing mission, the principal vehicle needs of the more scientifically oriented *Prospector* mission would be automatically provided. Before industrial studies of this problem could be initiated, it is essential that the *C-3* system concept and the performance capabilities be established on a reasonably firm basis.

In conclusion it must be noted that, in evaluating systems for carrying out the manned lunar-landing mission, attention must be given to the continuing manned space-flight missions which will carry on after the attainment of the initial goal. At this time, it appears that the continuing manned space-flight program will be associated with the operation of a continuously manned laboratory in a near-Earth orbit, with a similar operation on the lunar surface and, ultimately, with similar operations on the surface of one or more of the other planets or their satellites. The continuing nature of the manned space-flight program does not appear to modify strongly the criteria for selecting the final system, or systems, for implementation of the lunar-landing mission; however, the expected scale of resulting activity has some influence on the over-all criteria.

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